[redacted]

338

Z65-11428

CLASSIFICATION CHANGE

To UNCLASSIFIED

By authority of GDS - EO 11652 Date 12/31/72

Changed by L. Shirley

Classified Document Master Control Station, NASA
Scientific and Technical Information Facility

SID 62-338

Copy #4

TRAJECTORY ANALYSIS SUMMARY

PROJECT APOLLO (U)

15 March 1962 4.5.4.14

Prepared by

Performance and Trajectory Section

RECEIVED
MAR 20 1962
Technical Information Center
S&ID

~~This document contains information affecting the national defense of the United States within the meaning of the Espionage Laws, Title 18 U.S.C. Section 793 and 794. Its transmission or revelation of its contents in any manner to an unauthorized person is prohibited by law.~~

NORTH AMERICAN AVIATION, INC.
SPACE and INFORMATION SYSTEMS DIVISION

~~DOWNGRADED AT 3 YEAR~~
~~INTERVALS; DECLASSIFIED AFTER~~
~~12 YEARS DOD DIR 5200.10~~

[redacted]

FOREWORD

The "Trajectory Analysis Summary Report" is submitted in response to paragraph 4.5.4.14 of part 4 of the NASA Statement of Work, dated 19 December 1961.

[redacted]

CONTENTS

Paragraph	Title	Page
1.0	INTRODUCTION	1
2.0	NOMENCLATURE AND SYMBOLS	3
2.1	Apollo Flight Phase Definitions	3
2.1.1	Earth Orbital Mission	3
2.1.1.1	Ascent	3
2.1.1.2	Earth Orbit	3
2.1.1.3	Orbital Maneuvers	3
2.1.1.4	Orbit Ejection	3
2.1.1.5	De-orbit Coast	3
2.1.1.6	Entry	3
2.1.1.7	Recovery	4
2.1.2	Circumlunar Mission	4
2.1.2.1	Earth Orbit	4
2.1.2.2	Translunar Injection	4
2.1.2.3	Translunar Coast	4
2.1.2.4	Transearth Coast	5
2.1.3	Lunar Orbit Mission	5
2.1.3.1	Lunar Orbit Injection	5
2.1.3.2	Lunar Orbit	5
2.1.3.3	Transearth Injection	5
2.1.4	Lunar Landing Mission	5
2.1.4.1	Lunar Orbital Maneuver	5
2.1.4.2	Main Retro.	6
2.1.4.3	Lunar Landing	6
2.1.4.4	Lunar Ascent	6
2.2	Symbols	6
2.3	Subscripts	8
2.4	Definitions	8
3.0	FLIGHT PLAN CRITERIA	11
3.1	General	11
3.1.1	Launch Site	11
3.1.2	Landing Sites	11
3.1.3	Flight Time Capability	11
3.1.4	Entry and Recovery	11
3.2	Earth Orbital Mission (Phase A)	11
3.2.1	Recovery Requirements From Orbit	11

[redacted]

[redacted]

Paragraph	Title vv	Page
3.2.2	Orbital Altitudes	11
3.2.3	Orbital Inclination	11
3.2.4	Launch Azimuths	12
3.2.5	Injection Altitude	12
3.2.6	Orbital Maneuvers	12
3.2.7	Orbit Ejection	12
3.2.8	Atmospheric Entry	12
3.2.9	Recovery and Landing	12
3.3	Circumlunar Mission (Phase B)	12
3.3.1	Launch Flexibility	12
3.3.2	Perilune Altitude	13
3.3.3	Parking Orbit	13
3.3.4	Launch Azimuths	13
3.3.5	Translunar Midcourse Corrections . .	13
3.3.6	Translunar Insertion Position . . .	13
3.3.7	Transearth Orbit Inclination . . .	13
3.3.8	Transearth Midcourse Corrections . .	13
3.3.9	Entry	13
3.3.10	Recovery and Landing	14
3.4	Lunar Orbit Mission (Phase B)	14
3.4.1	Translunar	14
3.4.2	Lunar Orbit	14
3.4.2.1	Propulsion	14
3.4.2.2	Orbit Duration	14
3.4.3	Transearth	14
3.4.3.1	Injection Conditions	14
3.4.4	Propulsion	15
3.4.5	Entry	15
3.5	Lunar Landing Missions (Phase C) . .	15
3.5.1	Earth Orbit Rendezvous	15
3.5.2	Lunar Orbital Rendezvous	15
3.5.3	Direct	15
3.5.3.1	Earth Parking Orbit	15
3.5.3.2	Translunar	15
3.5.3.3	Lunar Orbit	15
3.5.3.4	Lunar Landing	15
3.5.3.5	Lunar Boost	16
3.5.3.6	Transearth Injection	16
3.5.3.7	Transearth Midcoruse Corrections .	17
3.5.3.8	Entry	17
3.5.3.9	Recovery and Landing	17
4.0	REQUIREMENT ANALYSIS	19
4.1	Characteristic Velocities	19
4.2	Flight Plan Requirements	20
4.2.1	Earth Orbital Mission	20

[redacted]

[redacted]

Paragraph	Title	Page
4.2.1.1	Ascent	20
4.2.1.2	Orbit	22
4.2.1.3	Ejection	22
4.2.1.4	Entry	23
4.2.1.5	Recovery	28
4.2.2	Circumlunar Mission	28
4.2.2.1	Initial Phase	28
4.2.2.1.1	Ascent	28
4.2.2.1.2	Orbit	31
4.2.2.1.3	Mid-course phase	31
4.2.2.1.3.1	Injection	31
4.2.2.1.3.2	Coast	35
4.2.2.1.4	Terminal Phase	35
4.2.2.1.4.1	Entry	35
4.2.2.1.4.2	Range Considerations	39
4.2.2.1.4.2.1	Corridor	43
4.2.2.1.4.2.2	Emergency Mode	46
4.2.2.1.4.2.3	Recovery	46
4.2.3	Lunar Orbit Mission	46
4.2.3.1	Translunar	46
4.2.3.1.1	Initial Phase	46
4.2.3.1.1.1	Ascent	46
4.2.3.1.1.2	Orbit	46
4.2.3.1.2	Mid-course Phase	46
4.2.3.1.2.1	Injection	46
4.2.3.1.2.2	Coast	46
4.2.3.1.3	Terminal Phase	46
4.2.3.1.3.1	Injection	48
4.2.3.1.3.2	Orbit	48
4.2.3.2	Transearth	51
4.2.3.2.1	Initial Phase	51
4.2.3.2.2	Mid-course Phase	51
4.2.3.2.2.1	Injection	51
4.2.3.2.2.2	Coast	56
4.2.3.2.3	Terminal Phase	56
4.2.4	Lunar Landing Missions (Phase C)	56
4.2.4.1	Earth Orbital Rendezvous	56
4.2.4.2	Lunar Orbital Rendezvous	56
4.2.4.3	Direct	56
4.2.4.3.1	Translunar	56
4.2.4.3.1.1	Initial Phase	56
4.2.4.3.1.1.1	Ascent	56
4.2.4.3.1.1.2	Orbit	56

[redacted]

Paragraph	Title	Page
4.2.4.3.1.2	Mid-course Phase	56
4.2.4.3.1.2.1	Injection	56
4.2.4.3.1.2.2	Coast	56
4.2.4.3.1.3	Terminal Phase	57
4.2.4.3.1.3.1	Injection	57
4.2.4.3.1.3.2	Orbit	57
4.2.4.3.1.3.3	Descent	57
4.2.4.3.1.3.3.1	Lunar Orbital Transfer	57
4.2.4.3.1.3.3.2	Main Retro	57
4.2.4.3.1.3.3.3	Landing	57
4.2.4.3.2	Transearth	57
4.2.4.3.2.1	Initial Phase	62
4.2.4.3.2.1.1	Ascent	62
4.2.4.3.2.1.2	Orbit	62
4.2.4.3.2.2	Mid-course Phase	62
4.2.4.3.2.2.1	Injection	64
4.2.4.3.2.2.2	Coast	68
4.2.4.3.2.3	Terminal Phase	68
4.3	Abort Requirements	68
4.3.1	General	68
4.3.2	Typical Trajectories	68
4.3.2.1	Earth Orbital Mission	68
4.3.2.1.1	Ascent	68
4.3.2.1.2	Orbit	69
4.3.2.2	Lunar Missions	73
4.3.2.2.1	Translunar	73
4.3.2.2.1.1	Initial Phase	73
4.3.2.2.1.2	Mid-course Phase	74
4.3.2.2.2	Lunar Vicinity	74
4.3.2.2.2.1	Circumlunar	74
4.3.2.2.2.2	Lunar Orbit	74
4.3.2.2.2.3	Lunar Landing	74
4.3.2.2.3	Transearth	74
5.0	DESIGN TRAJECTORY CRITERIA	79
5.1	Nominal Trajectories	79
5.2	Off-Design Trajectories	79
5.3	Atmospheric Flight Phases	79
5.3.1	Boost	79
5.3.1.1	C-1 Launch Vehicle	81
5.3.1.1.1	S-I Stage	81
5.3.1.1.2	S-IV Stage	81
5.3.1.2	C-5 Launch Vehicle	81

Paragraph	Title	Page
5.3.1.2.1	S-IC Stage	82
5.3.1.2.2	S-II Stage	82
5.3.1.2.3	S-IVB Stage	82
5.3.2	Entry	82
5.3.2.1	Entry Corridor - Lunar Missions	82
5.3.2.1.1	Design Corridor Criteria	82
5.3.2.1.2	Operational Corridor Limits	83
5.3.2.1.3	Mid-course Guidance Corridor	83
6.0	DESIGN TRAJECTORIES	85
6.1	Structural Loads and Heating Trajectories	85
7.0	TRAJECTORY ENVELOPES	95
7.1	Mission Duration Envelopes	95
7.2	Launch and Ascent Envelopes	98
7.3	Space Operation Envelopes	98
7.4	Entry Envelopes	98
7.4.1	Entry Range	98

[redacted]

ILLUSTRATIONS

Figure	Title	Page
3-1	Transit Time Effects on Gross Weight	18
4-1	Saturn C-1 Orbital Payload for Direct Ascent	21
4-2	Required Lateral Range at Landing	24
4-3	Performance Envelopes - Orbital Entry	26
4-4	Ejection Velocity Characteristics	27
4-5	Launch Azimuth vs Relative Launch Time	29
4-6	Lunar Injection Loci	33
4-7	Circumlunar Trajectory Geometry	36
4-8	Operational and Entry Requirements	37
4-9	Inertial Range Envelopes Entry Conditions	41
4-10	Corridor Characteristics	44
4-11	Effect of Bank Angle on Corridor	45
4-12	Emergency Mode Characteristics	47
4-13	Lunar Orbit Injection Requirements	49
4-14	Lunar Orbit Injection Geometry	50
4-15	Lunar Satellite Perturbations	52
4-16	Lunar Satellite Perturbations	53
4-17	Velocity Increment Required for Trans-earth Injection	54
4-18	Trans-earth Injection Requirement Limits	55
4-19	Lunar Landing Mission Geometry	58
4-20	Landing Module Propulsion Requirements - Vertical Landing	59
4-21	Landing Module Propulsion Requirements - Horizontal Landing	60
4-22	Service Module Propulsion Requirements	61
4-23	Trans-earth Injection Requirement	63
4-24	Trans-earth Transit Time	65
4-25	Vertical Boost Launch Parameters	66
4-26	Vertical Boost Launch Parameters	67
4-27	Typical Pad Abort Trajectories	70
4-28	Typical Axial Acceleration Time Histories	71
4-29	Boost Abort Envelope	72
4-30	Coplanar Abort at Escape Velocity	75
4-31	Typical Class II Abort Impulsive Velocity Requirements	76
4-32	Abort Entry Limits	78
5-1	Altitude-Velocity Profiles	80
6-1	Design Loads Trajectory No. 1	87
6-2	Design Loads Trajectory No. 2	89
6-3	Design Loads Trajectory No. 3	91
6-4	Design Loads Trajectory No. 4	93

~~CONFIDENTIAL~~

TABLES

Table	Title	Page
1	Transit Time - Gross Weight Parameter Study	17
2	Mission Propulsion Requirements	19
3	Launch Vehicle Weight and Performance	22
4	Landing Site Locations	73
5	Design Trajectories Summary	85
6	Earth Orbital Mission Duration	95
7	Circumlunar Mission Duration	95
8	Lunar Orbit Mission Duration	96
9	Lunar Landing Mission Duration	97

CO[redacted]

1.0 INTRODUCTION

This report provides a summary of the pertinent trajectory analysis studies that explain the "how and why" of various flight plan criteria for all Apollo missions. In addition, the design trajectory criteria will be included and the pertinent design trajectories presented for the various systems such as structures, heat shield, guidance and navigation systems, and environmental control systems. This document will be revised periodically as more detail is included in the requirements analyses.

[redacted]

CO[redacted]

2.0 NOMENCLATURE AND SYMBOLS

2.1 Apollo Flight Phase Definitions. - The definition of the Apollo flight phases are presented in sequence for the earth orbital and the lunar missions. Flight phases common to more than one mission are so indicated and are not repeated.

2.1.1 Earth Orbital Mission. -

2.1.1.1 Ascent. - Ascent defines the period between launch vehicle liftoff and earth orbit injection. The ascent phase includes:

1st Stage Boost

1st Stage Separation

2nd Stage Boost Into Earth Orbit

The ascent phase is common to all missions.

2.1.1.2 Earth Orbit. - Earth orbit begins at the termination of second stage burning and ends at the initiation of the orbit ejection phase. The earth orbit phase includes the separation of the Apollo Spacecraft from the launch vehicle and all spacecraft attitude control maneuvers.

2.1.1.3 Orbital Maneuvers. - The orbital maneuver flight phase shall be used to change the orbital elements or to simulate rendezvous.

2.1.1.4 Orbit Ejection. - Defines the retro powered flight phase required to place the spacecraft on a de-orbit coast trajectory.

2.1.1.5 De-Orbit Coast. - De-orbit coast follows orbit ejection, continues through the vacuum descent from orbit and terminates at the entry interface. The command module is separated from the service module and oriented for entry during this phase.

2.1.1.6 Entry. - Entry begins at the entry interface (400,000 feet) and ends at the recovery interface altitude (100,000 feet). The entry phase includes:

(a) Pull-up to horizontal flight

(b) Transition from Pull-up to Glide

(c) Glide to Recovery Interface

The entry phase is common to all missions.

[redacted]

2.1.1.7 Recovery. - Recovery begins at the recovery interface (100,000 feet) and ends at touchdown. The recovery phase includes:

(a) Glide to Drogue Chute deployment

(b) Drogue Chute deployment and descent

(c) Main Chute deployment and descent

The recovery phase is common to all missions.

2.1.2 Circumlunar Mission. -

2.1.2.1 Earth Orbit. - For the lunar missions the earth orbit is used to place the spacecraft and the launch vehicle in the inertial position required for translunar injection. Included in this phase are:

(a) Second Stage Separation

(b) Coast Attitude Control

This phase is common to all lunar missions.

2.1.2.2 Translunar Injection. - Translunar injection begins at the start of the powered firing sequence of the launch vehicle final stage and terminates at launch vehicle engine shutdown. The velocity vector at shutdown will yield a free return circumlunar trajectory. Included are:

(a) Firing Sequence prior to Engine Ignition

(b) Launch Vehicle Stage Boost

This phase is common to all lunar missions.

2.1.2.3 Translunar Coast. - Translunar Coast follows translunar injection and terminates at perilune. This phase includes:

(a) Separation of the Launch Vehicle from the Spacecraft

(b) Required mid-course velocity corrections provided by the Service Propulsion System. (The lunar landing module propulsion for the landing mission)

(c) Attitude Control Maneuvers

This phase is common to all lunar missions.

2.1.2.4 Transearth Coast. - Transearth coast commences at perilune for the circumlunar mission and at transearth injection for the lunar orbit and the lunar landing mission. This phase is common to all lunar missions and includes:

(a) Required mid-course velocity corrections provided by the Service Propulsion System

(b) Attitude control maneuvers

(c) Separation of the command module from the service module after the final mid-course velocity correction

(d) Orientation of the command module for entry.

2.1.3 Lunar Orbit Mission. -

2.1.3.1 Lunar Orbit Injection. - The Lunar Orbit Injection is a retro power phase initiated at perilune to reduce hyperbolic approach velocity to lunar orbital velocity.

NOTE: The Service Propulsion System provides the retro velocity for the lunar orbit mission and the Landing Module Propulsion for the landing mission. This phase includes the firing sequence, prior to engine ignition, and is common to both the lunar orbit and landing missions.

2.1.3.2 Lunar Orbit. - The lunar orbit follows lunar orbit injection for both the orbit mission and the landing mission. In the case of the landing mission it follows the lunar ascent phase as well. The lunar orbit is used for lunar surveillance and for proper positioning of the spacecraft for lunar landing or transearth injection. Specifically the orbits used for the landing mission, following lunar orbit injection or lunar ascent, are called the "approach orbit" and "departure orbit" respectively. This phase includes all attitude control maneuvers and is common to both the lunar orbit and the landing missions.

2.1.3.3 Transearth Injection. - The transearth injection is a powered phase, that follows lunar orbit and utilizes the Service Propulsion System to provide the velocity vector at shutdown required to satisfy entry constants.

2.1.4 Lunar Landing Mission. -

2.1.4.1 Lunar Orbital Transfer. - The lunar orbital transfer is a short retro powered phase from circular lunar orbit, utilizing the Landing Module Propulsion followed by a coasting 180° descent, to place the spacecraft in position for a lunar landing approach. This transfer may be used for lunar surveillance on orbit missions with the service module providing the propulsion. For this case the spacecraft shall be allowed to coast back to its original altitude prior to applying a small velocity increment that re-establishes a circular orbit. This phase includes firing sequences prior to engine ignition and attitude control maneuvers during coasting flight.

2.1.4.2 Main Retro. - The main retro phase starts at perilune of the Hohmann transfer and terminates at 1,000 ft. above the lunar surface. The landing module propulsion system is used to slow the spacecraft to almost a zero velocity while steering it to a position approximately 1,000 ft above the landing site. This phase includes the firing sequence prior to engine ignition.

2.1.4.3 Lunar Landing. - The landing phase commences at approximately 1,000 ft and terminates at lunar touchdown. It includes hover, site selection, translation, and touchdown. The lunar landing is accomplished by using the landing module propulsion system.

2.1.4.4 Lunar Ascent. - Lunar ascent begins at lunar liftoff and terminates at injection into lunar orbit. Lunar ascent is accomplished by using the Service Propulsion System.

2.2 Symbols. -

e	eccentricity
g	acceleration or deceleration in units of earth g
GHA	Greenwich hour angle, deg.
G	load factor
h	altitude, ft, n. miles
i, ι	inclination angle, deg.
I, I_{sp}	specific impulse, sec.
L/D	lift-to-drag ratio
M	Mach number
$\bar{q}$	dynamic pressure, psf.
R	radius, ft, n. miles. Also range, n. miles

T	thrust, lbs.
t	time
u	node-to-vehicle angle, deg.
V	velocity (inertial reference except during atmospheric flight), fps.
W	weight of subject vehicle, lbs.
W/C_DS	entry weight-drag parameter, lb/sq ft.
WRT	with respect to . . .
β	in-plane angle, range angle, deg.
δ	out-of-plane angle (relative inclination), deg.
ϵ	angle of abort engine thrust vector relative to x-axis
γ	flight path angle relative to local horizontal, deg.
$\Delta\Omega$	angular regression of node (due to earth's bulge), deg.
ω	argument of perifocus, deg.
$\Delta\omega$	angular advance of perigee (due to earth's bulge), deg.
Φ	declination of moon, deg.
ϕ	vehicle bank angle (during entry), deg.
ρ	lateral range angle, deg.
Σ	azimuth (clockwise from north), deg.
Ⓗ	right ascension of the moon, deg.
θ	body attitude, longitude, deg.

☊	ascending node, deg.
☋	descending node, deg.

2.3 Subscripts. -

C	characteristic, coast
E, EN, RE	entry, re-entry
I	injection, ejection
L	launch
M, m	lunar, moon
o	initial, sea level
ORB	orbit
p	perifocus, perilune, perigee
TE	transearth
V	vertical
∞	in a vacuum environment, at infinity

2.4 Definitions. -

Apogee	The point on a geocentric elliptical orbit farthest from the center of the earth.
Azimuth	Angle from the north meridian to the normal projection of the velocity vector on the local horizontal plane. N-E-S-W rotation.
Celestial Sphere	A hypothetical sphere of infinite radius centered at the observer and referenced to the axis of rotation of the earth and the vernal equinox.
Declination	Angle between the position vector and its normal projection on a reference plane.

Elevation	Angle between the velocity vector and its normal projection on the local horizontal plane.
Geocentric	Referenced to the center of the earth.
Inclination	Angle (in a plane normal to the line of nodes) between the orbit plane and a reference plane.
Lunicentric	Referenced to the center of the moon.
Nodes	The points of intersection on a reference sphere of two great circle planes.
Perifocus	The point on an orbit nearest the dynamical center.
Perigee	The point on a geocentric orbit nearest the center of the earth.
Perilune	The point on a lunicentric orbit nearest the center of the moon.
Right Ascension	Angle between the normal projection of the position vector on a reference plane and a reference axis in the reference plane.

3.0 FLIGHT PLAN CRITERIA

3.1 General. - The following flight plan criteria shall apply to all manned Apollo missions.

3.1.1 Launch Site. - The launch site shall be Cape Canaveral, Florida. The launch azimuths shall be within range safety requirements.

3.1.2 Landing Sites. - The primary landing site shall be at a latitude of 29.48 degrees N and longitude of 99.00 degrees W (San Antonio, Texas). The secondary site shall be at a latitude of 29.48 degrees S and longitude of 135.00 degrees E (Woomera, Australia).

3.1.3 Flight Time Capability. - Spacecraft systems shall be capable of performing at their nominal design performance level for a mission of 14 days without resupply. For lunar landing missions, 7 of the 14 days may be spent on the lunar surface.

3.1.4 Entry and Recovery. - The spacecraft shall have the capability of initiating an entry and landing maneuver at any time during any mission. This is qualified to the extent that spacecraft propulsion limitations may require a time delay prior to initiating abort for return to a specific landing site. Abort analyses of each mission shall determine the abort delay time as a result of available characteristic velocity.

3.2 Earth Orbital Mission (Phase A). -

3.2.1 Recovery Requirements From Orbit. - The spacecraft shall be capable of landing at the primary site (or the secondary site) from three consecutive orbits per day. Sufficient alternate sites shall be designated to provide recovery capability from any single orbit. This necessitates a minimum number of six sites located 60 degrees apart along the north or south parallels corresponding to the latitude of the primary site. The final number will depend upon geographical and political considerations.

3.2.2 Orbital Altitudes. - The spacecraft systems shall be capable of performing at their normal design performance level within orbital altitude limits of 90 nautical miles to 400 nautical miles. The orbital flight time requirement of 14 days (maximum) will be modified in the lower altitude levels to correspond to natural orbital lifetime limits.

3.2.3 Orbital Inclination. - Orbital inclination shall be selected on the basis of minimum ejection ΔV requirements that satisfy the recovery (three consecutive passes) requirements for the latitude of the selected landing site. Since the optimum inclination is also related to orbit altitude, the design inclinations shall be based on the maximum and minimum attainable orbital altitude. The orbital inclination and subsequent entry and recovery trajectory shall define GOSS track requirements. However, these requirements shall be compatible with existing tracking stations.

3.2.4 Launch Azimuths. - The required launch azimuths from Cape Canaveral are determined by orbital inclination and range safety.

3.2.5 Injection Altitude. - The maximum allowable injection altitude at orbit injection shall be defined by a direct ascent optimum boost trajectory (maximum payload) having the following constraint: The combination, velocity-altitude-path angle, at any time during boost shall be consistent with those flight conditions defining the safe abort corridor. The allowable entry conditions for abort are defined by a 20 g load factor. The minimum allowable injection altitude shall be defined by the maximum (3σ) guaranteed orbital altitude attainable with a single engine out (from lift-off) of the S-I stage (C-1 launch vehicle).

3.2.6 Orbital Maneuvers. - Spacecraft systems shall be capable of performing orbital maneuvers of sufficient duration and magnitude for systems checkout, midcourse corrections and mission objectives that may require maneuvers.

3.2.7 Orbit Ejection. - The predicted spacecraft position at orbit ejection shall be defined by the time taken between ejection and the recovery interface altitude corresponding to a preselected trajectory. The ejection maneuver requirements (thrust vector orientation and magnitude) shall be determined by the following criteria: (1) orbital altitudes between 90 and 400 nautical miles, (2) maximum out-of-plane touchdown range, (3) maximum aerodynamics maneuver envelope, and (4) minimum guidance sensitivity.

3.2.8 Atmospheric Entry. - The nominal entry trajectory associated with the predicted ejection position point shall be based upon the range associated with an equilibrium glide trajectory corresponding to an L/D of 0.35. Lateral range shall nominally be accomplished by out-of-plane retro at orbit ejection.

3.2.9 Recovery and Landing. - The recovery interface altitude shall be designated at 100,000 feet. The touchdown dispersion envelope shall be ±_____ nautical miles.

3.3 Circumlunar Mission (Phase B). -

3.3.1 Launch Flexibility. - The circumlunar flight plan shall contain sufficient flexibility to permit a launch period of at least ten consecutive launch days out of each month. This period will be centered about the maximum positive and negative lunar declinations. The flight plan shall include at least a two-hour period on the launch date in which the mission can be launched either continuously or at discrete intervals. The circumlunar trajectory shall be designed on the basis of a free-return transearth trajectory which returns to the primary landing site within the GOSS envelope. Arrival time corrections are not considered a deviation of this criteria.

3.3.2 Perilune Altitude. - The nominal design perilune altitude will vary with launch date to obtain timing flexibility for return to one of the primary sites. The allowable variation in perilune altitude will be established by the landing mission propulsion limits.

3.3.3 Parking Orbit. - The minimum time in earth parking orbit shall not be less than that required to checkout the spacecraft and improve the guidance accuracy with navigation sightings. The maximum time may be as long as required to satisfy total mission timing for return to a primary landing site. The upper limit should be approximately 4 orbits. The parking orbit altitude shall be dictated by maximum payload and orbit lifetime.

3.3.4 Launch Azimuths. - The launch direction from Cape Canaveral shall be determined by lunar declination and limited to azimuths dictated by range safety.

3.3.5 Translunar Midcourse Corrections. - The midcourse navigation correction requirements including the error in arrival velocity shall not exceed 500 fps.

3.3.6 Translunar Insertion Position. - Final insertion into the translunar trajectory shall be located such that the trajectory can be determined by GOSS within 15 minutes following translunar insertion burnout. Insertion burnout position shall be pre-determined prior to each mission based upon lunar declination and launch delays up to the maximum.

3.3.7 Transearth Orbit Inclination. - The inclination of the approach conic shall be determined by control of arrival time within the GOSS earth traces. The inclination will be approximately 30 degrees to the earth's equator.

3.3.8 Transearth Midcourse Corrections. - The midcourse navigation correction requirements including the error in arrival velocity shall not exceed 500 fps.

3.3.9 Entry. - The location of the spacecraft at the entry interface relative to the preselected landing site shall nominally depend upon the lunar declination, launch delay, and inclination of the approach conic. For (1) two primary sites, (2) a two hour launch window, (3) ten day launch flexibility about the minimum and maximum declination, and (4) fixed earth track, the required entry range shall be 3000 nautical miles to 5500 nautical miles. The command module systems shall be designed to include these values of range plus sufficient range tolerances to account for landing site contingencies, such as bad weather, for all entry conditions within the operational corridor.

3.3.10 Recovery and Landing. - Spacecraft systems shall be compatible with water or land recoveries. Emergency nighttime landing capabilities shall be required.

3.4 Lunar Orbit Mission (Phase B). -

3.4.1 Translunar. - See Circumlunar mission.

3.4.2 Lunar Orbit. -

3.4.2.1 Propulsion. - Injection into a circular orbit of a specified altitude will occur at perilune of a circumlunar trajectory. The propulsion (velocity impulse) required to inject into the orbit will be established from a circumlunar trajectory having the following characteristics:

(1) The moon will be at apogee at the time of perilune

(2) The inclination of the translunar trajectory plane to the moon's orbit plane will be maximum

(3) The inclination of the transearth trajectory plane to the moon's orbit plane will be maximum.

The propulsion shall allow for a 5-degree out-of-plane maneuver at injection.

3.4.2.2 Orbit Duration. - The minimum time spent in orbit will be one circular orbit period at the specified orbit altitude. The maximum time in orbit will be such that the total mission time does not exceed 14 days. The maximum time in orbit for a specific mission will be reduced, if necessary, by the criteria that the recovery phase of the transearth trajectory will occur in daylight.

3.4.3 Transearth. -

3.4.3.1 Injection Conditions. - Injection into the transearth trajectory will be possible at least once during each revolution in orbit. Requirements of the transearth trajectory will determine the exact position in the orbit and the corresponding time at injection.

3.4.4. Propulsion. - The propulsion requirements for the transearth injection will be based upon the following criteria. The inclination of the circular orbit plane to the moon's orbit plane will be 15 degrees, and the line of nodes will have the most unfavorable orientation from the standpoint of minimizing the injection velocity. The moon will be at apogee at the time of injection and the inclination of the transearth trajectory plane to the moon's orbit plane will be maximum. A 24-hour return flight time flexibility will be maintained to permit recovery at a specific earth site.

3.4.5 Entry. - Entry will be within the ground tracking corridor and entry range limits defined for the circumlunar mission. Maximum and minimum flight times will result from the establishment of the propulsion requirements for transearth injection.

3.5 Lunar Landing Missions (Phase C). -

3.5.1 Earth Orbital Rendezvous. - No data available at present

3.5.2 Lunar Orbital Rendezvous. - No data available at present

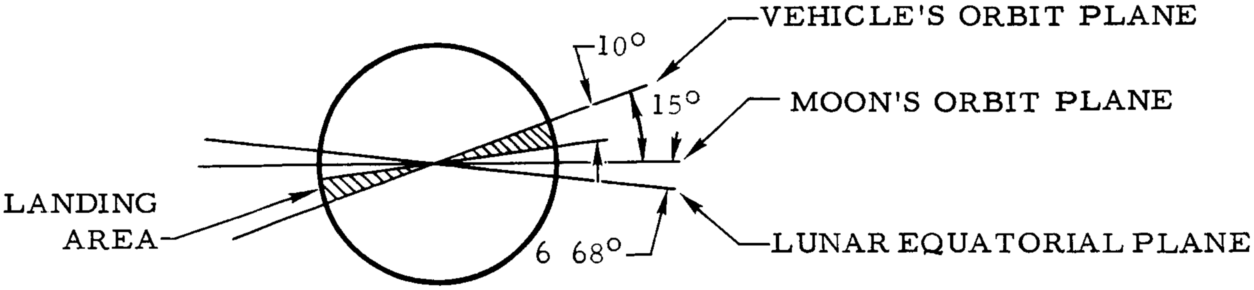
3.5.3 Direct. -

3.5.3.1 Earth Parking Orbit. - The vehicle shall be injected into an earth parking orbit from the Atlantic Missile Range (AMR) to obtain launch flexibility. The boost to orbit will be accomplished using a NOVA.

3.5.3.2 Translunar. - The translunar trajectory shall be of the "free return" type and shall satisfy all criteria specified for the circumlunar mission.

3.5.3.3 Lunar Orbit. - The vehicle shall inject into a circular lunar orbit at translunar perilune with a planar angle change capability of up to 5 degrees. The limits on the orientation of the orbit plane will be dictated by the requirement that the translunar trajectory be of the "free return" type and the 5 degrees planar angle change capability. At least one pass will be made over the vicinity of the landing site prior to initiating the landing phase. The parking orbit altitude will be dictated by the translunar "free return" trajectory.

3.5.3.4 Lunar Landing. - The landing phase will be initiated by making a Hohmann transfer to a perilune altitude which is compatible with the guidance capability and the terminal maneuver requirements. The landing site must be located such that the plane of the parking orbit will contain the landing site at the time of touchdown. The inclination restraints imposed by the translunar trajectory requirements and the 5 degree planar angle change capability require the landing site to be in the vicinity of the moon's orbital plane. The maximum latitude allowable (selenographic latitude) will be 21.68 degrees and can be reached only if specific geometric relationships exist. The line of nodes of the lunar equatorial plane and the selenocentric translunar conic relative to the moon's orbit plane must coincide, and the landing site must be ninety degrees from the line of nodes as shown in the following sketch.



The specific landing area will depend upon the precise date of translunar perilune.

3.5.3.5 Lunar Boost. - The vehicle will be boosted into a lunar parking orbit prior to injection into the transearth trajectory. The parking orbit will be used to provide a uniform boost profile regardless of the launch site location and minimize the variation in propulsion requirements. Sufficient launch azimuth flexibility must be available to properly orient the parking orbit plane for injection into the transearth trajectory without a planar angle change. The launch will be into a direct orbit relative to the moon's rotation. The injection into the transearth trajectory will then be visible from the earth.

3.5.3.6 Transearth Injection. - The vehicle will nominally inject into a transearth trajectory which returns along the northern boundary of the GOSS range when the entry range angle is a minimum. Launch delays can be obtained without varying the transearth injection velocity by varying the inclination of the geocentric conic. As launch delays occur, the launch parameters are varied such that the inclination of the geocentric conic is varied and the entry range angle increases. The allowable launch window is dependent upon the allowable inclination flexibility and the entry range flexibility.

The minimum injection velocity would be determined by the maximum allowable transit time. The upper limit on transit time is approximately 90 hours. Beyond this point, the trajectories become very sensitive and their geometry deviates greatly from earth conic solutions. The resultant trajectories would be difficult to program in the guidance computer. Also, beyond 90 hours the injection velocity must increase in order to maintain the proper entry conditions. A primary consideration should be to minimize the Spacecraft gross weight during normal and abort situations. Table 1 illustrates the effects on gross weight as the transit time is decreased.

Table 1.- Transit Time-Gross Weight Parameter Study

Parameter	Gross Weight Effect
Guidance	decrease
Main Propulsion	increase
Life Support System	decrease
General Radiation Shielding	decrease
Entry Heat Shielding	increase
Solar Flare Protection	decrease

If the solar flare prediction techniques are found to be reliable, the advance notice period could establish the minimum transit time if the shielding requirements become prohibitive. The effects described above are illustrated in Figure 3.1. If solar flare shielding is incorporated in the command module design, the minimum transit time will be determined by the flight time-earth landing site location compatibility. For a single landing site, a twenty-four hour variation in the transit time is required with a fixed ground track. The transit time flexibility can be reduced to twelve hours by using the alternate landing sited proposed for the earth orbital missions and an inclination variation equivalent to a four-hour delay.

3.5.3.7 Transearth Mid-Course Corrections. - Same as 3.3.8

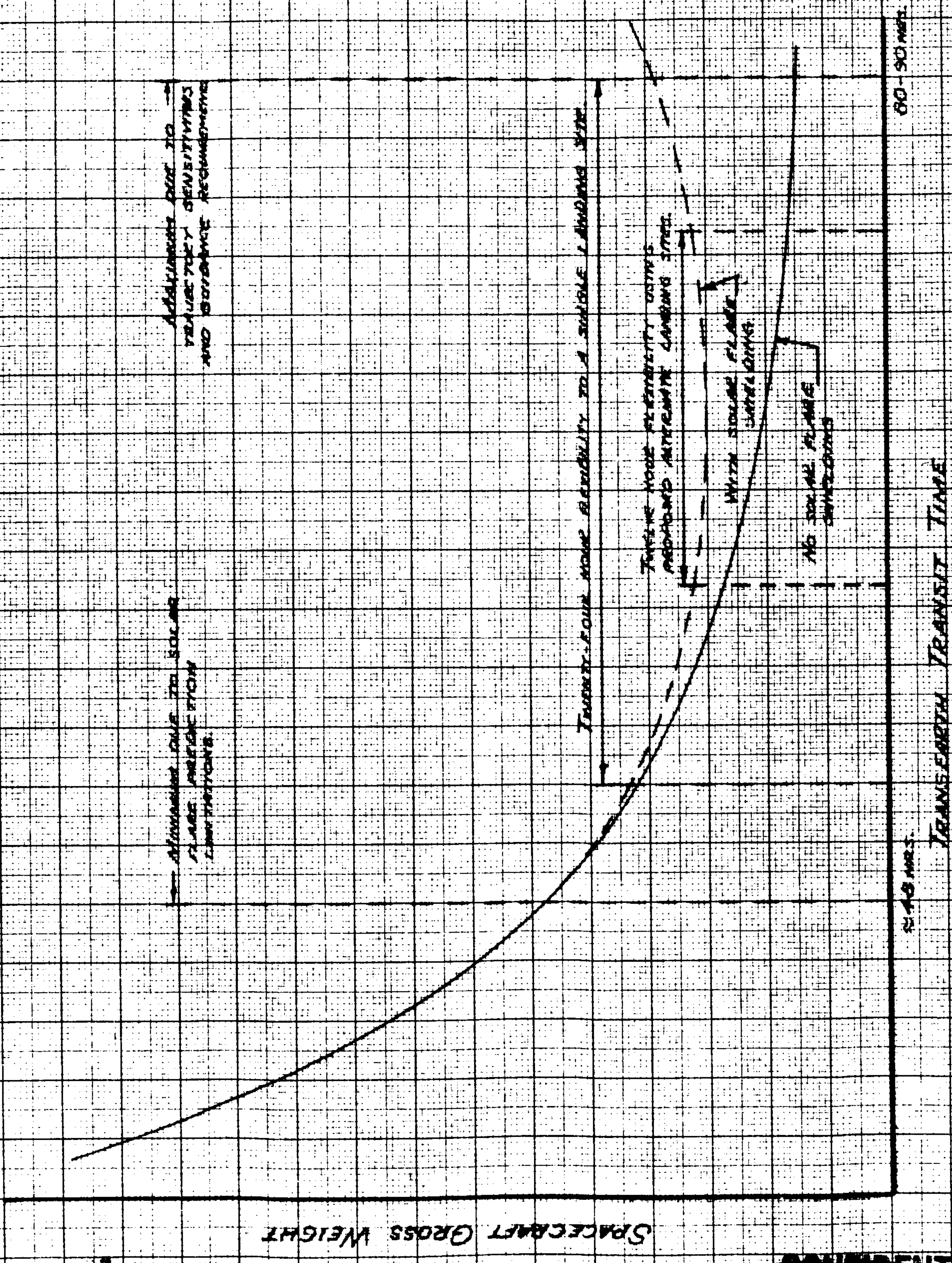
3.5.3.8 Entry. - Same as 3.4.5

3.5.3.9 Recovery and Landing. - Same as 3.3.10

Figure 3-1

[redacted]

TRANSIT TIME EFFECTS ON GROSS WEIGHT



[redacted]

4.0 REQUIREMENT ANALYSIS

4.1 Characteristic Velocities. - Examination of the Apollo Spacecraft missions has established the following performance requirement. Table 2 summarizes the characteristic velocities associated with these requirements.

Table 2. Mission Propulsion Requirements

Mission	Phase	Characteristic Velocity (FPS)
Earth Orbital	Orbital Transfer	665
	Ejection From Orbit	1640
		2305
Circumlunar	Mid-Course Translunar	500
	Mid-Course Transearth	500
		1000*
Lunar Orbit	Mid-Course Translunar	500
	Injection Into Orbit	3380
	Injection Out of Orbit	4820
	Mid-Course Transearth	500
		9200
Lunar Landing	Translunar	
	Mid-Course	500
	Injection	3380
	Hohmann Transfer	115
	Main Retro	5778
	Hover and Landing	1000
		10773
	Transearth	
	Boost	5980
	Injection	3355
	Mid-Course	500
	Tolerances and Control	500
		10335

*10,000 FPS Preliminary Abort Value

For the earth orbital missions, the orbital transfer velocity is based on a Hohmann transfer from 200 to 400 nautical miles. The ejection velocity reflects de-orbit from 400 nautical miles and includes a lateral maneuver at ejection that results in a cross-range of 163 nautical miles at touchdown. The lunar mission injection values are based on free return type approaches to the moon that are inclined 10 degrees to the lunar orbit plane. Velocities include a 5 degree planar angle change. For the lunar orbit mission the transearth injection velocity provides for a 24 hour variation in transit time and is based on returning to the earth along a trajectory inclined 63 degrees to the lunar orbit plane. The hover and landing allowance of 1000 fps is preliminary. For all spacecraft lunar missions the propellant tanks shall be fully fueled.

4.2 Flight Plan Requirements. -

4.2.1 Earth Orbital Mission (Phase A). -

4.2.1.1 Ascent. - The Apollo Spacecraft will be launched with a C-1 booster into a nominal 200 nautical mile circular orbit. In the event that one engine should fail during the C-1 ascent, the remaining engines must be capable of delivering the spacecraft into a circular orbit not less than 126 nautical miles. The payload limit data curve of the C-1 booster is presented in Figure 4-1, from which it may be seen that an Apollo Spacecraft weighing 18,966 pounds could be delivered into a 200 nautical mile orbit. Present abort recovery limitations preclude a boost into orbit that terminates in excess of 200 nautical miles. Selection of operational orbital altitudes below 200 nautical miles is the prerogative of the command center and is limited only by the acceptable orbital decay. Figure 4-1 presents payload weights which include the Apollo command module, service module, and spacecraft adapter. The payloads are based upon trajectories using a due east launch from AMR. The other trajectory criteria assumptions are enumerated as follows and in Table 3. The S-1 stage burnout and jettison weight includes the Apollo launch escape system and S-I/S-IV separation and start propellants. The S-IV stage burnout and jettison weight includes the 3,000-pound instrument and boost guidance section. Thrust and specific impulse have been adjusted for nozzle cant angles and inerts lost during stage burning. It is assumed the S-I consumes part of the listed available propellant during a 3-second mainstage hold down period prior to vehicle release. The minimum S-IV stage specific impulse has been used to compensate for wind profiles, launch azimuth deviations, and guidance disturbances which have not been included in the computer performance curves.

Ejection from orbit and landing require that the nominal orbital inclination be 31.88 degrees. As a consequence, launch of the Apollo from

~~CONFIDENTIAL~~

PREPARED BY: Wlw

CHECKED BY:

DATE: 10 FEB 1962

NORTH AMERICAN AVIATION, INC.
MISSILE DIVISION

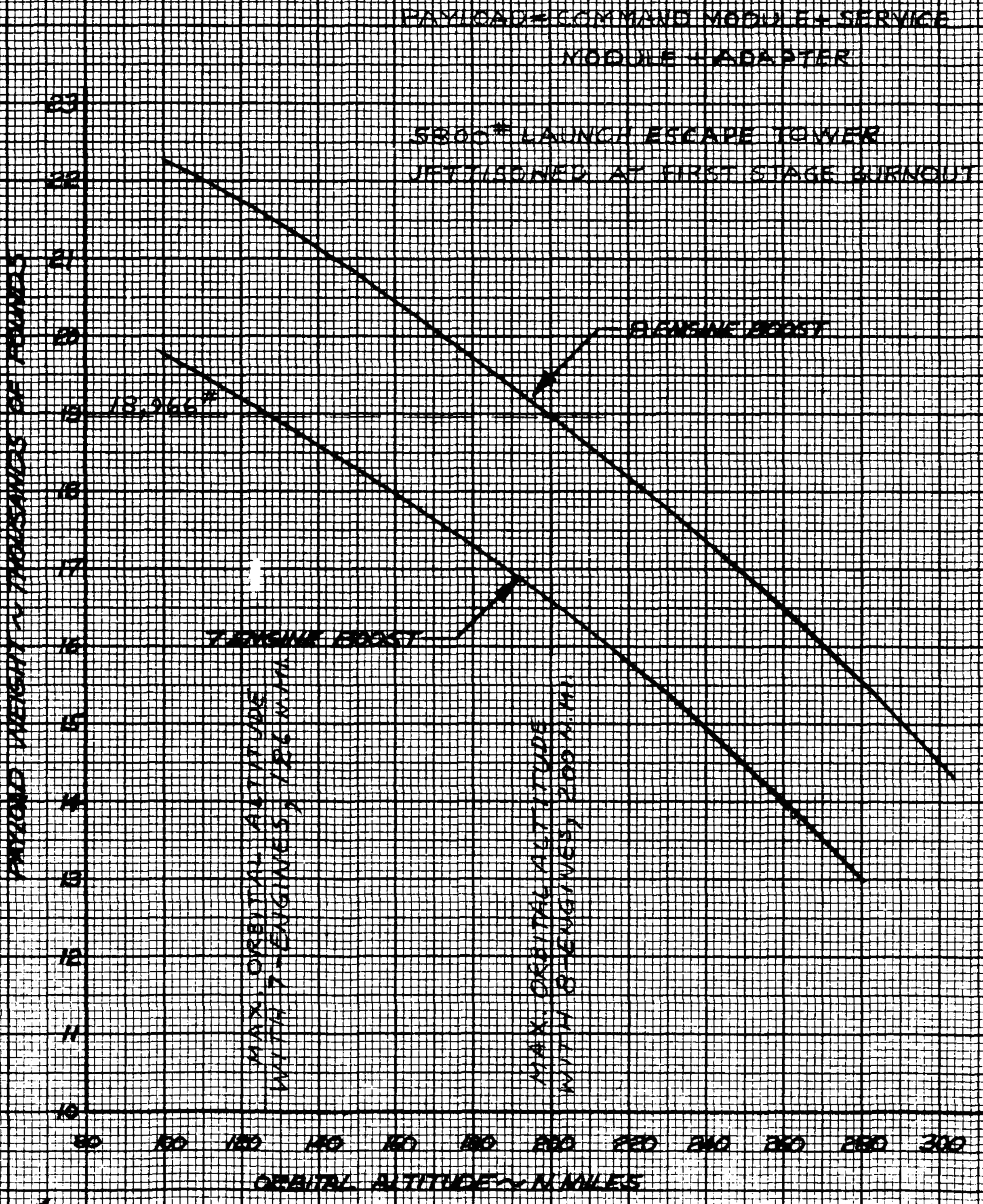
Figure 4-1

PAGE NO. 21 OF

REPORT NO. SID 62-338

MODEL NO.

SATURN C-I
ORBITAL PAYLOAD FOR DIRECT ASCENT



~~CONFIDENTIAL~~

Table 3. Launch Vehicle Weight and Performance

Item	Measurement	
S-1 Stage Gross Weight	971, 204	lbs.
S-1 Stage Usable Propellant	844, 900	lbs.
S-1 Stage Inerts Lost	2, 592	lbs.
S-1 Stage Burnout and Jettison Weight	123, 712	lbs.
S-1 Stage Specific Impulse (Sea Level)	253.85	sec.
S-1 Stage 8-Engine Thrust (Sea Level)	1, 498, 850	lbs.
S-IV Stage Gross Weight	115, 742	lbs.
S-IV Stage Usable Propellant	99, 500	lbs.
S-IV Stage Inerts Lost	60	lbs.
S-IV Stage Burnout and Jettison Weight	16, 182	lbs.
S-IV Stage Specific Impulse (Vacuum)	409.50	sec.
S-IV Stage 6-Engine Thrust (Vacuum)	89, 507	lbs.

Cape Canaveral requires that boost be terminated in an inertial plane with azimuthal bearing at Cape Canaveral at either 75 degrees or 105 degrees.

4.2.1.2 Orbit. - The design maximum time for orbiting is fourteen days. An orbital altitude in excess of 139 nautical miles is necessary to meet this requirement. Orbital altitudes from 90 to 126 nautical miles may be satisfied by direct ascent of a seven-engine C-1 configuration, 90 to 200 nautical miles with an eight-engine C-1 configuration, and 200 to 400 nautical miles by an orbital transfer utilizing the service module propulsion system.

4.2.1.3 Ejection. - The spacecraft shall be capable of landing at the primary landing site (or at the backup site) from at least three orbits per day. In addition, alternate sites which may involve either land or water landing will be designated such that at least one alternate site can be reached for a landing from each orbit. The most efficient way to meet this requirement is to make the three recoveries consecutive.

C[redacted]

Presented in Figure 4-2 is general data of required maximum lateral range at landing to satisfy the three orbital pass requirement. It may be seen that the selection of the landing site as San Antonio, Texas, (Latitude 29.48 degrees N) and a nominal orbital altitude of 250 nautical miles will require a maximum lateral range at landing (ρ) of 2.4 degrees, with a corresponding orbital inclination of 31.88 degrees. When the Apollo spacecraft is ejected from orbit other than 250 nautical miles, a slight variation in orbital inclination would be optimum.

The ejection window capability and the associated requirements are valid for all landing sites located on the north and south 29.48 degree parallels. A minimum of six landing sites located along either parallel and spaced 60 degrees apart in longitude would satisfy the above ejection window requirements. However, two of these sites, namely, India and Libya, are geographically and politically impractical. As a consequence, sites in the southern hemisphere must be substituted. In making these 180 degree-phase position substitutions, it will be noted that a seventh site in the southern hemisphere must be added to satisfy the full ejection window requirement. This provides a reserve of coverage which permits the desirable substitution of the Woomera site for a mid-Atlantic site in the northern hemisphere. Thus a total of seven sites, three in the northern hemisphere and four in the southern hemisphere represents a practical minimum number of sites.

4.2.1.4 Entry. - The combined ejection, coast, and atmospheric entry maneuver to the landing site shall be selected with regard to the following conditions:

(1) Maximum aerodynamic maneuverability

(2) Minimum ejection propulsion

(3) Minimum range sensitivity.

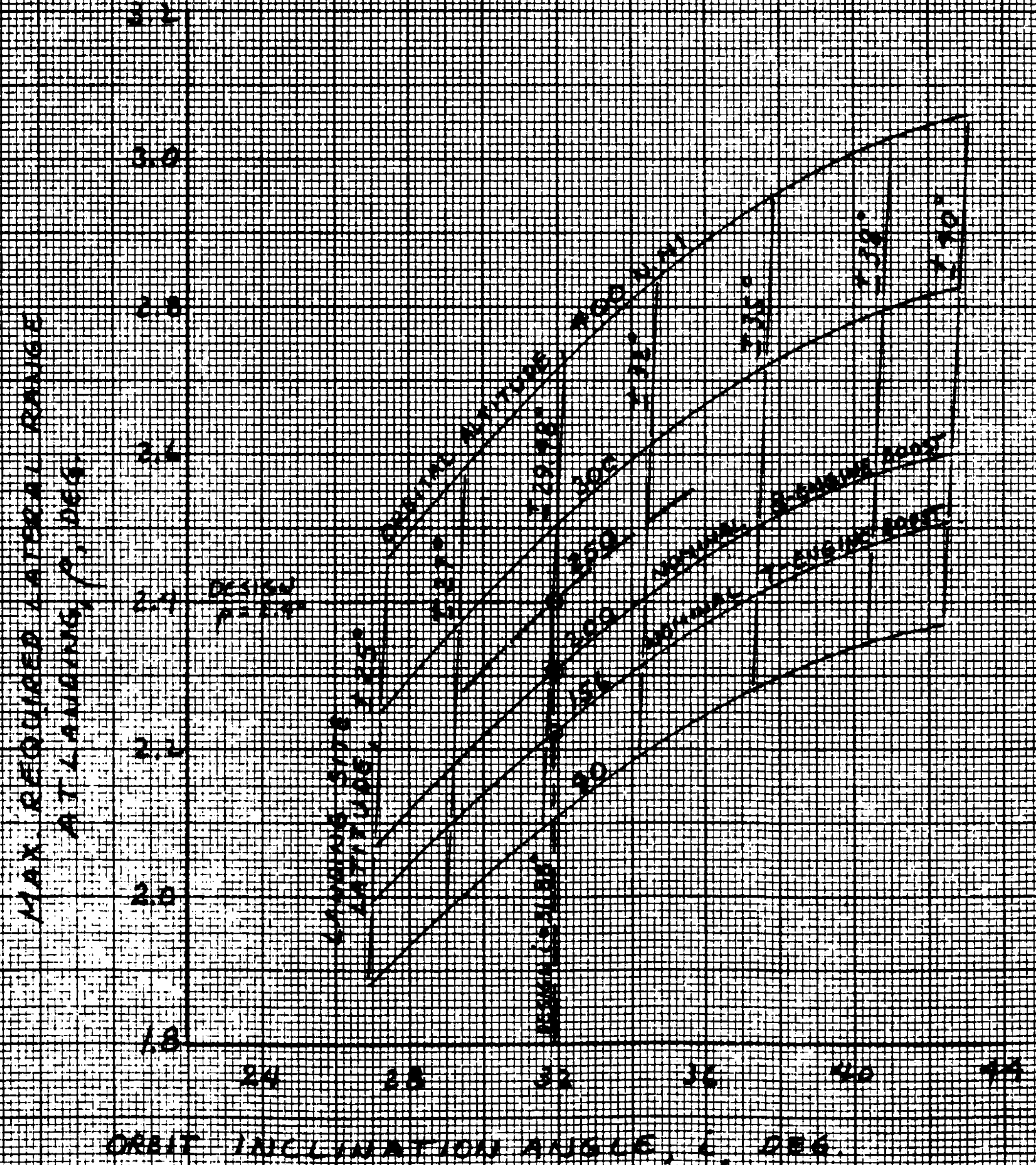
Maximum aerodynamic maneuver capability occurs as the combination of entry path angle and velocity approaches the skip-out boundary. The required maneuver capability, however, shall depend upon how accurately the entry position, velocity, time and path angle can be controlled. Errors in these parameters will be caused by errors in the magnitude of the ejection velocity, errors in the alignment of the ejection velocity, errors in determining the orbit parameters, and errors in the predicted time used to establish the orbit exit point. Since errors at entry must be removed by the range control system, guidance sensing tolerances must be within the vehicle's aerodynamic maneuver capability. Guidance sensing tolerances, however are related to propulsion characteristics in that range sensitivity due to errors in the magnitude of the imparted velocity (ΔV) decreases with increasing ejection ΔV. Further, increasing the magnitude of imparted

[redacted]

PREPARED BY: Dcw	**NORTH AMERICAN AVIATION, INC.** SPACE and INFORMATION SYSTEMS DIVISION	PAGE NO. 24 OF
CHECKED BY:		REPORT NO. ~~SID~~ 62-338 ~~CONFIDENTIAL~~
DATE: 2-5-62	Figure 4-2	

REQUIRED LATERAL RANGE AT LANDING

THREE CONSECUTIVE ORBITAL PASSES



~~CONFIDENTIAL~~

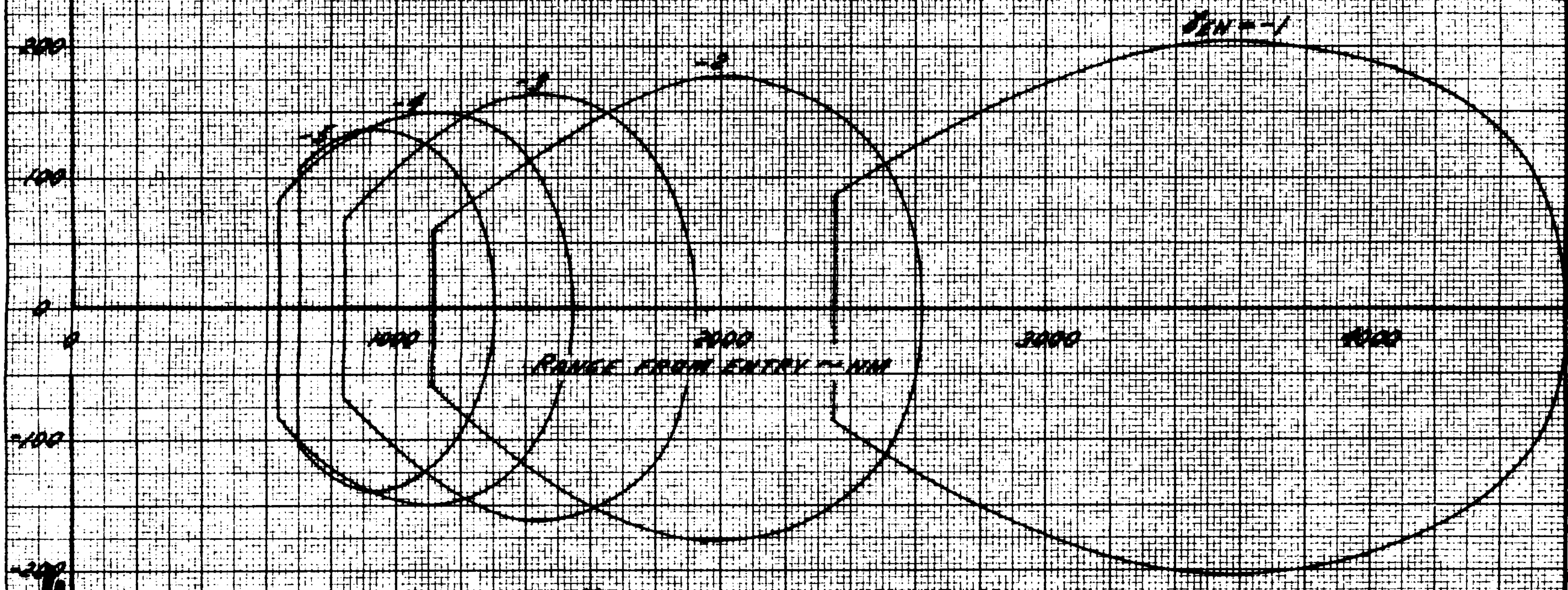
velocity results in decreasing the available range control during entry as well as adding vehicle weight penalties. Thus, the interrelations between propulsion, guidance, and vehicle control must be evaluated before final ejection requirements can be established. The aerodynamic maneuver capability is illustrated in Figure 4-3 for several entry conditions. The entry conditions correspond to ejection from a circular orbit of 156 nautical miles. Each entry corresponds to a specific ejection ΔV. For each ΔV the ejection angle was aligned to correspond to the angle for minimum range sensitivity, i.e. $\partial \beta / \partial \delta = 0$. The resulting envelopes are based upon $(L/D)_{max}$ of 0.5 and a 10g maximum load factor. The ejection velocity (ΔV) corresponding to these data is indicated in Figure 4-4 (curve labeled $\partial \beta / \partial \delta = 0$). The magnitudes of ejection ΔV includes an out-of-plane ejection maneuver that results in a cross-range angle of 2.22 degrees at touchdown (See Figure 4-2). The remaining data in Figure 4-4 illustrates how ejection requirements vary with specific entry conditions. For each entry glide range and corresponding entry path angle, an optimum total range for minimum propulsion exists ($\partial \Delta V / \partial \beta = 0$). These data are approximate in that glide range was assumed invariant with entry velocity for a given path angle and vehicle L/D. This approximation is reasonable however, particularly for small entry angles which correspond to the primary area of interest.

Thus, for minimum propulsion, the optimum range angle occurs at 98 degrees. The required ΔV at this range is 1080 fps. This corresponds to an entry path of two degrees and a nominal glide range of 31.7 degrees when entry is made at L/D of .35. The range control capability is approximately ±700 nautical miles (Figure 4-3) for a 10g limit and maximum $L/D = 0.50$. The range sensitivity with regard to propulsion ΔV errors is, however, greatest at this condition. As the total range angle is reduced for a specific entry condition, the range sensitivity is reduced at the expense of increased propulsion. Thus the optimum combination will correspond to range angles slightly less than those for minimum energy. The entry path angle will be slightly steeper than the overshoot boundary. The orbital inclination (31.88 degrees) in combination with the maximum lateral range (2.4 degrees) and a total range angle of approximately 90 degrees from ejection to landing results in a band of entry conics with inclinations between 29.48 degrees to 33.99 degrees. Examination of the geographic area encompassed by these conics to a given landing site on the 29.48 degrees parallel establishes the required entry track. The entry track is bounded by an inertial spherical triangle whose sides are formed by two conics of the same inclination (33.99 degrees), which intersect the landing site at approach azimuths of 72.3 degrees and 107.7 degrees, and whose base is an arc of a great circle centered on the landing site. (See Figure 4-4.) This inertial triangle when translated to geographic coordinates for location of ground tracking equipment must be corrected for earth rotation corresponding to the entry flight history with the smallest and largest transit time anticipated.

Figure 4-3

PERFORMANCE ENVELOPES - ORBITAL ENTRY

h_{ORBIT} = 150 NM
L/D = 0.5
W/C_DA = 50
G_{MAX} = 10



γ_{EN}	X_{TOTAL}	X_{CENTER} ϕ = 45°	V_{EN}
1	9800	3500	25785
2	5450	1950	25615
3	3625	1400	25395
4	2650	1100	25095
5	2100	920	24870

~~CONFIDENTIAL~~

PREPARED BY:

CHECKED BY

DATE:

NORTH AMERICAN AVIATION, INC.
SPACE and INFORMATION SYSTEMS DIVISION

Figure 4-4

PAGE NO. 27 OF

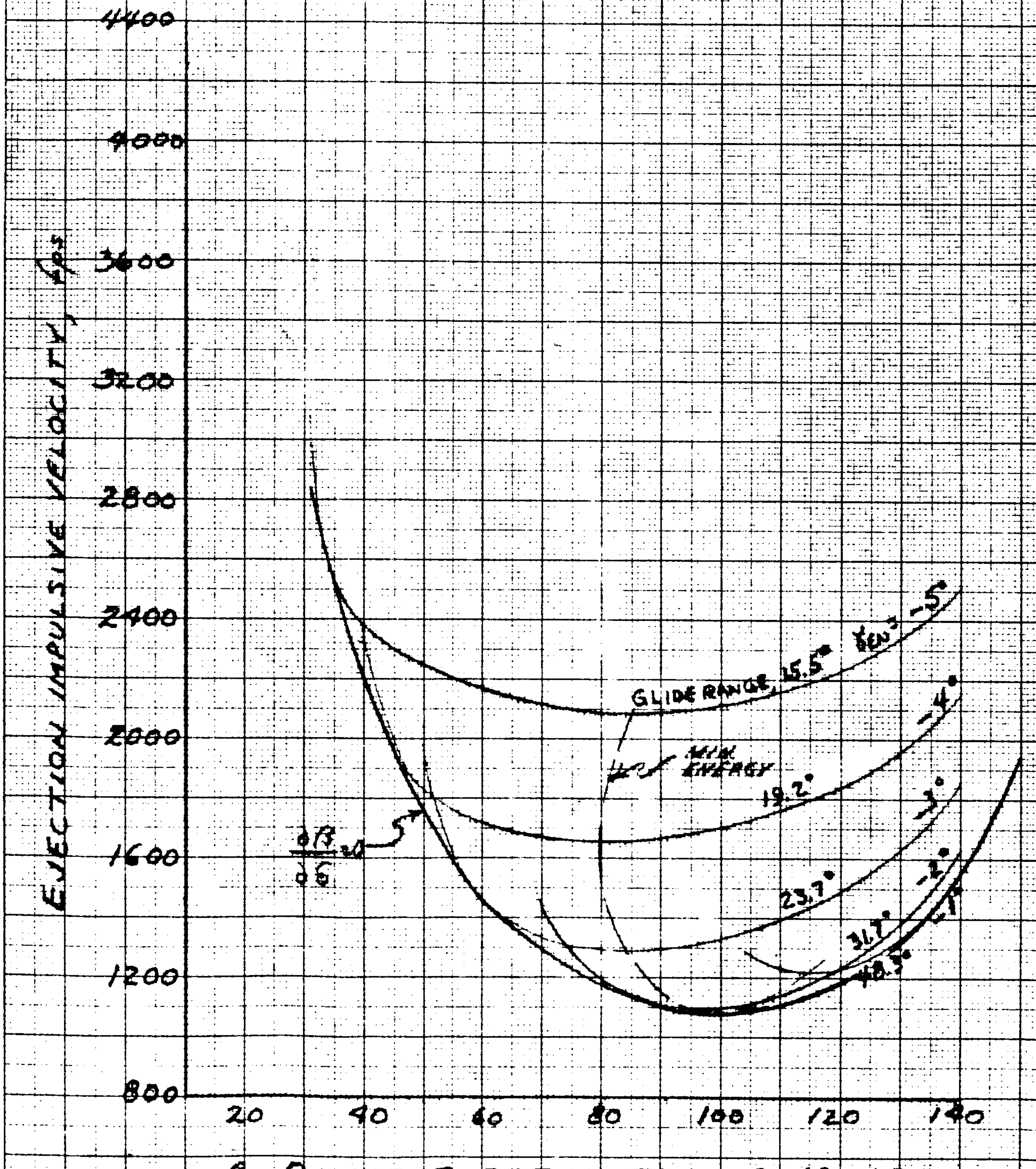
REPORT NO. SID 62-338

MODEL NO.

EJECTION VELOCITY CHARACTERISTICS

h_{ORBIT} = 150 N.M

γ_{EO} = 2.22° DEG.



The use of orbital inclinations steeper than 31.88 degrees (associated with altitudes above 250 nautical miles) or the ejection from orbit at ranges greater than 90 degrees from landing will expand the required GOSS coverage.

4.2.1.5 Recovery. - The interface between entry and recovery will be 100,000 feet. The command module will include an earth landing system to be used to satisfy earth landing requirements. The landing system may utilize parachutes or moderate L/D devices such as "parawing" or glide parachutes.

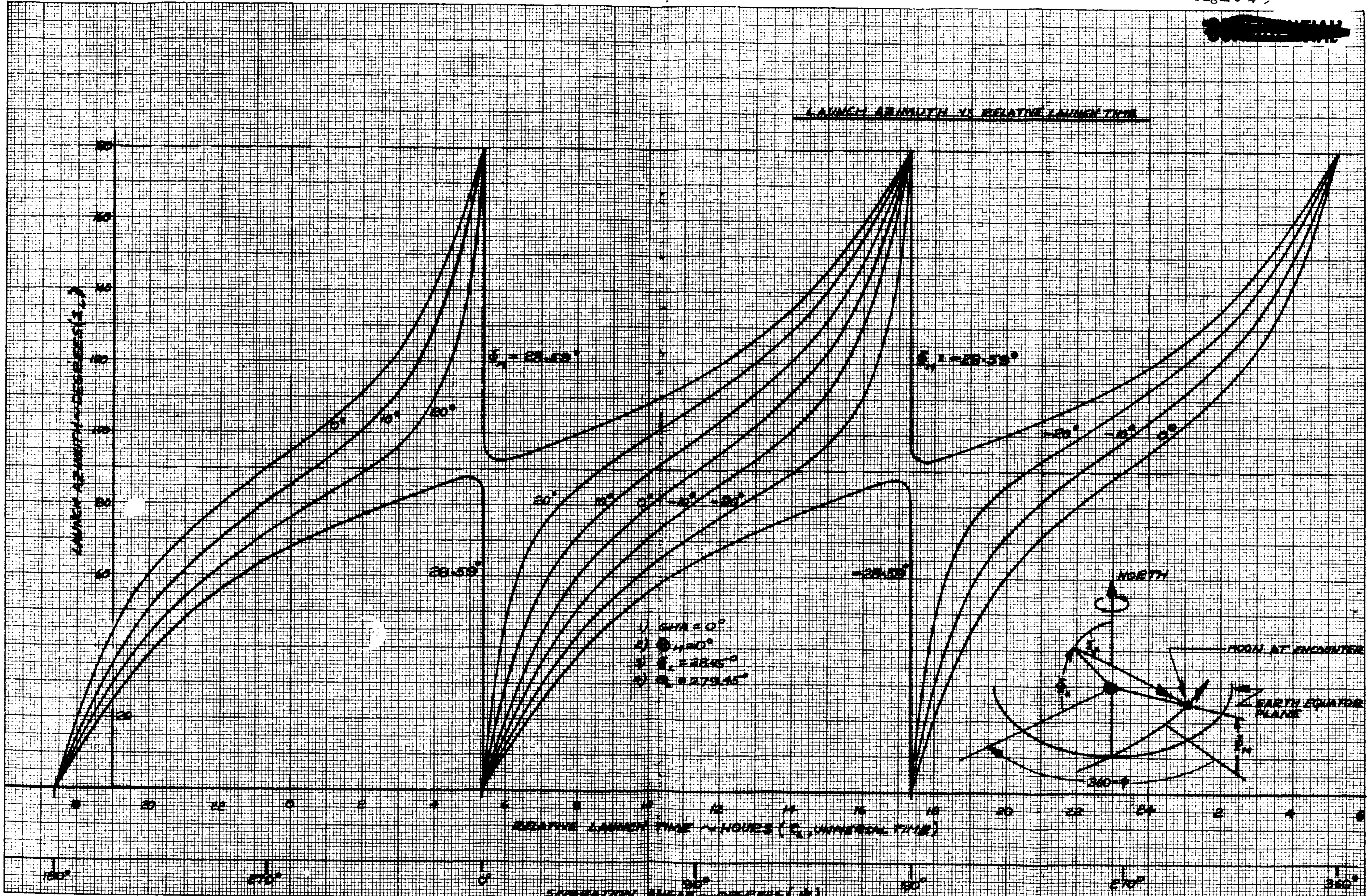
4.2.2 Circumlunar Mission. - The flight plan objectives in the circumlunar mission are: (1) to launch the spacecraft from a specified site, (2) to establish a circumlunar trajectory having a specified perilune altitude, and (3) to return the spacecraft safely to a selected landing site along a specified entry track.

Factors influencing the establishment of flight plan requirements for this mission are grossly inter-related. The predominant factor is the manner in which the mission flight time is made compatible with the landing site location on specific launch dates. Several possible methods for obtaining this compatibility are: (1) adequate launch azimuth variations, (2) wide GOSS earth trace limits, (3) the use of multiple parking orbits, (4) variation of the perilune altitude for specific launch dates, and (5) use of mid-course corrections to change arrival time. These methods are considered under the appropriate mission phases in the following discussion of the circumlunar mission flight requirements.

4.2.2.1 Initial Phase. - The initial phase of the mission is defined as that portion from launch up to translunar injection and includes the powered flight ascent and the parking orbit coast.

4.2.2.1.1 Ascent. - Circumlunar missions will originate at Cape Canaveral, Florida. An interval of ten consecutive days which yield launches satisfying the objectives of the circumlunar mission will be required each month. Also, a minimum launch delay capability of two hours is required. The geometric factors to be considered in the ascent include the effect on launch azimuth limits of launch delay capability and flight-time-landing site location compatibility. Figure 4-5 shows a typical variation of launch azimuth with time of day. The minimum allowable spread in launch azimuth, determined by the two hour launch delay capability, is ±7.5 degrees from due east. During the last six months of 1968 and all of 1969, when the moon is at maximum or minimum declination, due-east launches are not possible. On these dates a launch spread of about ±10 degrees from due east will provide a discontinuous two hour delay capability. In general, launch azimuth spreads which are not centered about the due-east direction may also be

[redacted]



LAUNCH AZIMUTH VS RELATIVE LAUNCH TIME

[redacted]

C[redacted]

greater than ±7.5 degrees. A variation of ±10 degrees is sufficient for the two hour delay capability in all cases

Launch azimuth variation can also be employed to obtain compatibility of total flight time and landing site locations. Range safety azimuth limits at the Cape (±24 degrees) allow about a 5.5 hour launch time variation of which two hours must be conserved for a possible delay. The remaining 3.5 hours could be used to alleviate the mission-time landing-site-location matching problem. Although Figure 4-5 shows two possible launch intervals each day for circumlunar flight only one of these intervals will, in general, satisfy the requirements for a free return to a specific landing. The two intervals correspond to a short parking orbit coast and a long parking orbit coast.

4.2.2.1.2 Orbit. - The parking orbit altitude is 100 nautical miles. Figure 4-6 shows a ground trace from launch to translunar injection for launch azimuths varying from 75 to 105 degrees. When the moon is at negative declinations, the long coast solution will be required for returns to San Antonio, Texas, and injection will occur over the Pacific Ocean near the western coast of the United States. When the moon is at positive declinations, the return is to Woomera, Australia, which requires that the injection point be west of Australia, in the vicinity where the azimuth curves intersect. When the moon is near zero declination the return may be either to San Antonio, or Woomera and the corresponding injection points lie over the mid-Pacific and the west coast of Africa, respectively.

It may be desirable to use the first half of the parking orbit for systems checkouts or to obtain tracking data. The short coast angles could be increased by one revolution to accommodate this procedure with a resulting limit of 0.5 to 1.5 parking orbit revolutions. The use of multiple parking orbits is a possible method for obtaining flight time-landing site location compatibility. This technique is restricted, however, due to the incremental rather than continuous nature of the flexibility achieved.

4.2.2.1.3 Mid-course Phase. - The mid-course portion of the circumlunar mission includes the translunar injection and the circumlunar coast.

4.2.2.1.3.1 Injection. - The location of translunar injection points is directly related to the parking orbit coast duration and is discussed in Section 4.2.2.1.2. Injection velocity can be varied to control the transearth trajectory plane inclination and will change with launch delay and with time of month. A one (1) ft/sec variation in injection velocity will change the transearth plane inclination about 10 degrees.

[redacted]

4.2.2.1.3.2 Coast. - The coast phase of all circumlunar trajectories must be such that the specified altitude of closest approach at the moon is achieved and "free return" capability exists for a safe return to one of the primary landing sites. The transit time from translunar injection to entry is controlled primarily by the perilune altitude and will vary only from about 138 to 139 hours for a 100 nautical mile perilune.

Since the transit time remains nearly constant for a fixed perilune altitude and directly influences the timing problem at entry, it appears that a trade off study between flight time-landing site compatibility for the circumlunar mission and propulsion requirements for the landing mission as a function of perilune altitude should be considered. A variable perilune altitude for specific launch dates may be more desirable.

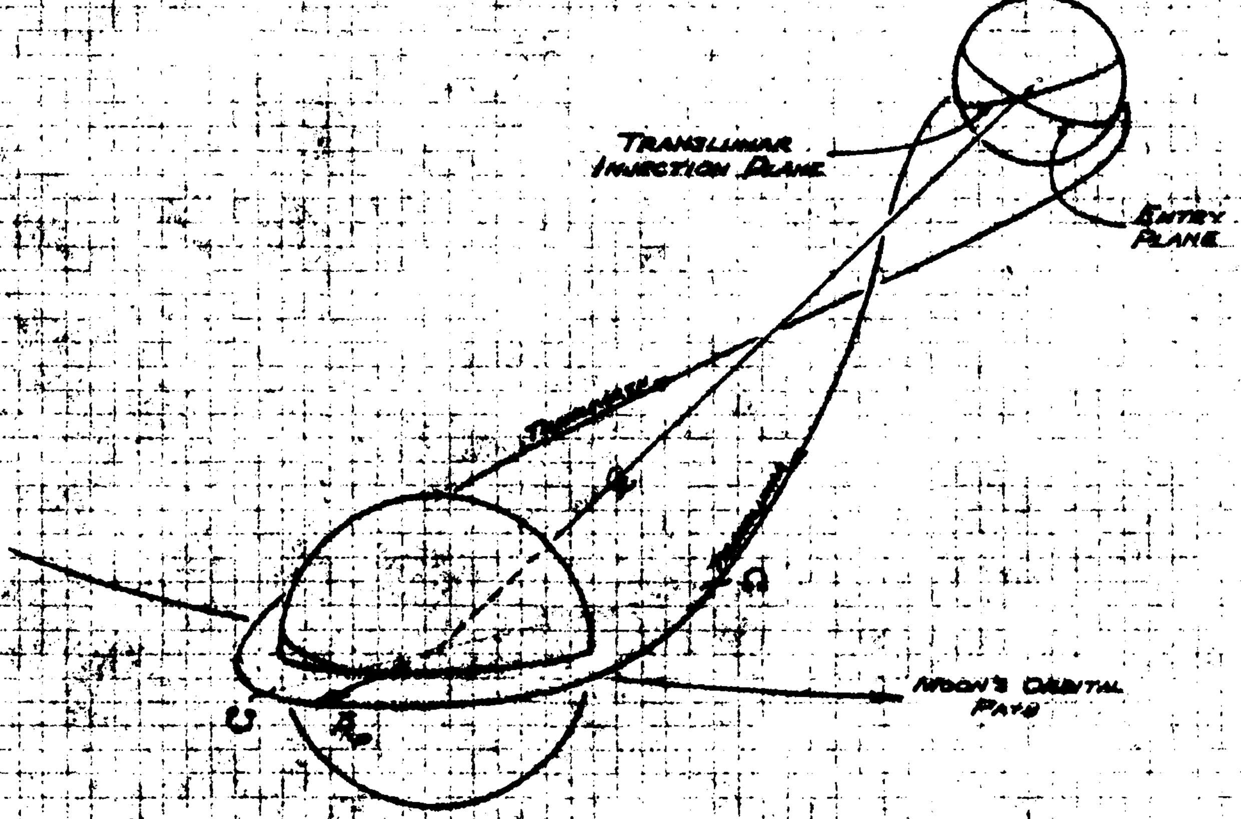
The coast phase exhibits certain aspects of symmetry when viewed in a coordinate system in which the reference plane contains the moon's orbit and the reference direction coincides with the moon's radius vector at the time of perilune. Inclination of the translunar plane to the moon's orbit plane in this system will vary from about 0 to 65 degrees. Inclination of the transearth trajectory plane to the moon's orbit plane will vary from 0 to 62.5 degrees depending on the landing site used (time of month). Only direct returns will be considered. Coast trajectories which originate on one side of the moon's orbit plane (either above or below) and return on the opposite side cross the plane only once in the lunar vicinity. Trajectories which originate and return on the same side of the moon's orbit plane cross the plane twice in the lunar vicinity. Figure 4-7 illustrates the latter case where both injection and entry occur below the moon's orbit plane.

4.2.2.1.4 Terminal Phase. - The terminal phase of the circumlunar mission includes the entry and recovery portions.

4.2.2.1.4.1 Entry. - Geometrical and dynamic flight plan requirements during entry pertain to (1) entry range limits, (2) timing, and (3) landing approach direction (transearth plane inclination limits). These factors are inter-related with each other and also influence the determination of launch azimuth limits discussed in Section 4.2.2.1.1. Circumulunar missions will terminate with a landing at San Antonio when the moon's declination at encounter is negative. The Woomera site will be utilized when the moon's declination at encounter is positive. Variation of the inclination of the transearth trajectory plane could be used to obtain launch time flexibility. Wide landing swaths, however, impose extreme entry range requirements and require extensive GOSS coverage capability. If some other technique is employed to obtain launch time flexibility the variation in transearth plane inclination can be restrained within the limits of GOSS track coverage established for the earth orbital mission, 29.48 to 33.99 degrees. Figure 4-8 shows the landing approach corridor to San Antonio, Texas, for this GOSS

Figure 4-7

CIRCUMLUNAR TRAJECTORY GEOMETRY


TRANSLUNAR INJECTION PLANE
ENTRY PLANE
℄
Ω
MOON'S ORBITAL PATH
℧
R_p

R_p ~ PERISELENIUM RADIUS VECTOR

℄ ~ EARTH-MOON LINE OF CENTERS
AT THE TIME OF PERISELENIUM

Ω ~ ASCENDING NODE OF THE SELENOCENTRIC CONIC

℧ ~ DESCENDING NODE OF THE SELENOCENTRIC CONIC

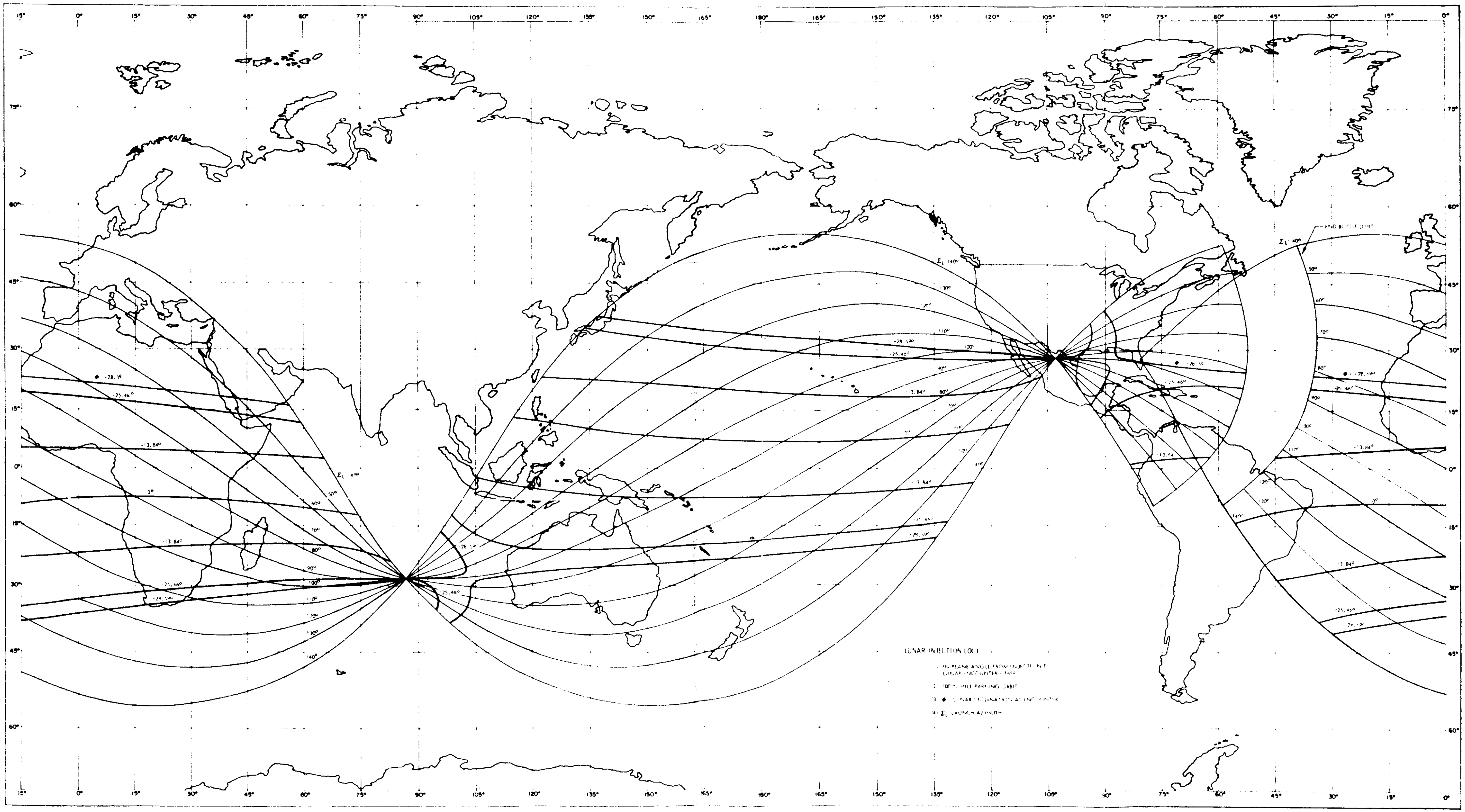
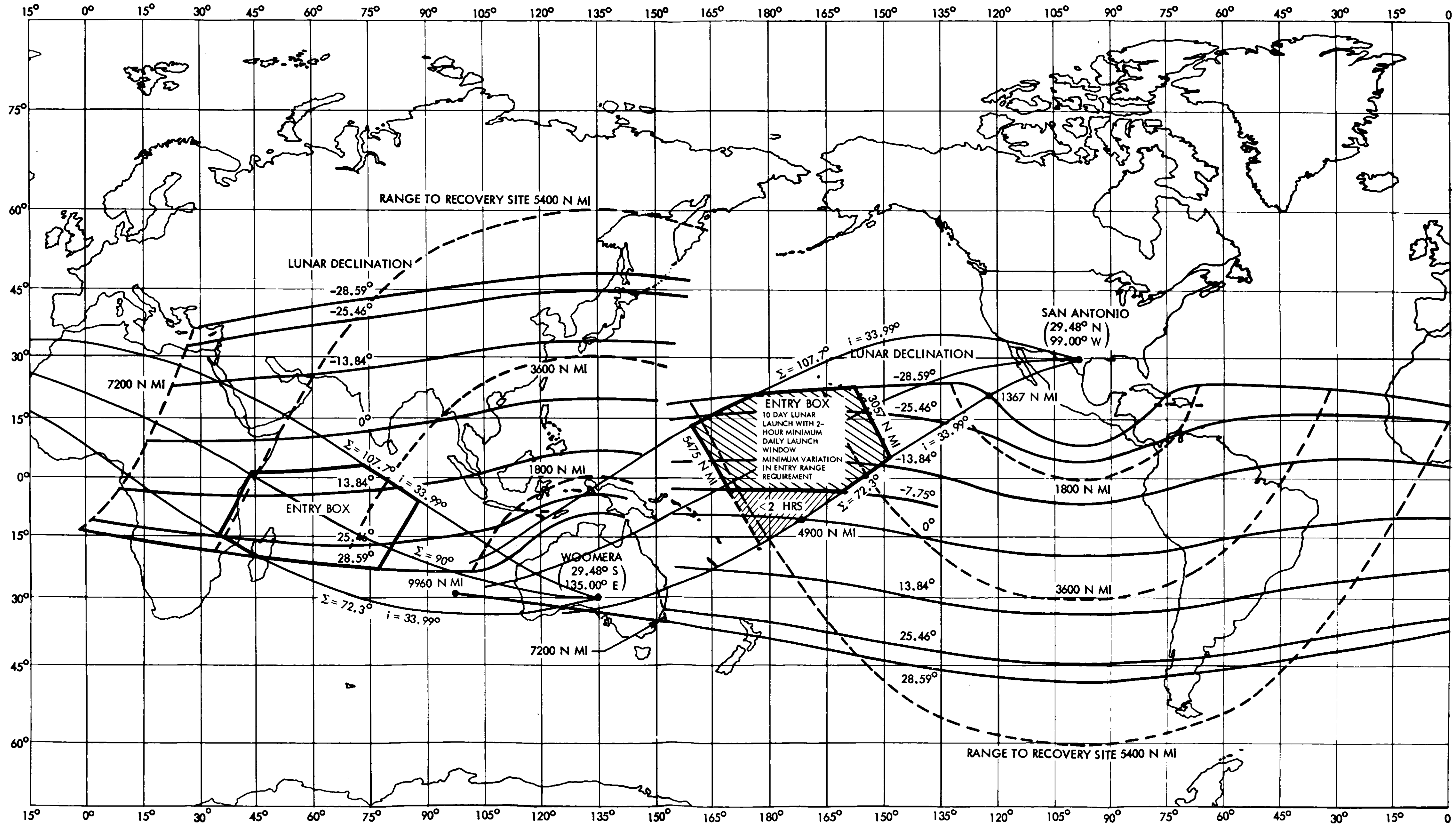

LUNAR INJECTION LOCI
1. IN-PLANE ANGLE FROM INJECTION TO LUNAR ENCOUNTER = 155°
2. FOR 100 N. MILE PARKING ORBIT
3. ◆ LUNAR DECLINATION AT ENCOUNTER
(4) Σ_L LAUNCH AZIMUTH
END BOOST LIMIT
Σ_L 40°
Σ_L 140°
-28.59°
-25.46°
-13.84°
0°
50°
60°
70°
80°
90°
100°
110°
120°
130°
140°
15°
0°
15°
30°
45°
60°
75°
90°
105°
120°
135°
150°
165°
180°
165°
150°
135°
120°
105°
90°
75°
60°
45°
30°
15°
0°
75°
60°
45°
30°
15°
0°
15°
30°
45°
60°

Figure 4-6. Lunar Injection Loci

[redacted]


15°
0°
15°
30°
45°
60°
75°
90°
105°
120°
135°
150°
165°
180°
165°
150°
135°
120°
105°
90°
75°
60°
45°
30°
15°
0
75°
60°
45°
30°
15°
0°
15°
30°
45°
60°
RANGE TO RECOVERY SITE 5400 N MI
LUNAR DECLINATION
-28.59°
-25.46°
-13.84°
0°
13.84°
25.46°
28.59°
7200 N MI
3600 N MI
1800 N MI
Σ = 107.7°
i = 33.99°
Σ = 90°
Σ = 72.3°
ENTRY BOX
9960 N MI
WOOMERA
(29.48° S
135.00° E)
7200 N MI
SAN ANTONIO
(29.48° N
99.00° W)
LUNAR DECLINATION
i = 33.99°
Σ = 107.7°
-28.59°
-25.46°
-13.84°
-7.75°
0°
13.84°
25.46°
28.59°
1367 N MI
3057 N MI
5475 N MI
4900 N MI
ENTRY BOX
10 DAY LUNAR LAUNCH WITH 2-HOUR MINIMUM DAILY LAUNCH WINDOW
MINIMUM VARIATION IN ENTRY RANGE REQUIREMENT
<2 HRS
Σ = 72.3°
1800 N MI
3600 N MI
RANGE TO RECOVERY SITE 5400 N MI

Operational and Entry Requirements

~~CONFIDENTIAL~~

track. A minimum variation in entry range requirements can be obtained by centering the ten consecutive launch dates about the time of minimum lunar declination. At minimum declination (-28.59 degrees) the nominal entry inclination is 29.48 degrees and the landing approach direction is due east. This condition establishes the minimum range requirement of 3,057 nautical miles. During a launch delay the inclination, approach azimuth and entry range all increase until after a two hour delay an inclination of 33.99 degrees is required. The maximum entry range required is established when the declination is about -7.75 degrees. The nominal entry inclination for these launches is 33.99 degrees and the approach direction is north of east. During launch delay the entry inclination decreases and the approach azimuth increases, until after a two hour delay the maximum range requirement of 5,475 nautical miles is established. Using 5,475 and 3,057 nautical miles as the entry range limits yields the entry box illustrated by the cross-hatched area in Figure 4-8 Launch delays greater than two hours are actually permissible for lunar declinations between -7.75 and -28.59 degrees. It will be desirable to have the earth landing occur during daylight. This means that of the 20 days possible for circumlunar missions only 10 will be available for daylight landings. In the worst case, only five consecutive days will be available for missions with daylight landing.

4.2.2.1.4.2 Range Considerations. - Within the launch and tracking constraints previously described, the guaranteed maximum range during the entry flight phase must be 5500 nautical miles. Since the spacecraft must achieve this range from all entry conditions, the critical entry from the standpoint of guaranteed range will be when the vehicle enters the atmosphere along the undershoot boundary. In addition, and without regard to the size of the corridor, the vehicle's realizable L/D is so sufficiently small that the required range can only be obtained by allowing some amount of exit from the atmosphere following the initial entry. The exit conditions and the distance traveled during the skip-out will depend upon the flight mode selected following the initial atmospheric penetration. Since no path control can be realized during skip-out, errors in the trajectory parameters at exit will result in range errors at re-entry. The magnitude of these errors must be within the range control limits of the vehicle during atmospheric re-entry. The problem is further complicated by the fact that not all entries will require maximum range. Thus, the guidance and control logic must be consistent for all range and entry conditions.

An inertial range envelope is shown in Figure 4-9 based upon a flight plan that maintained a constant initial pull-out altitude for each entry condition. When the vehicle had decelerated to a particular velocity, dependent upon the desired range, a roll maneuver was performed that resulted in a change in the vehicle's altitude and, thus, deceleration history. To define the footprint boundaries, i. e., maximum lateral range for a given longitudinal range, the vehicle was rolled to a bank angle of 45 degrees. The range variation was

FORM M-113-N-3 (REV. 5-61)

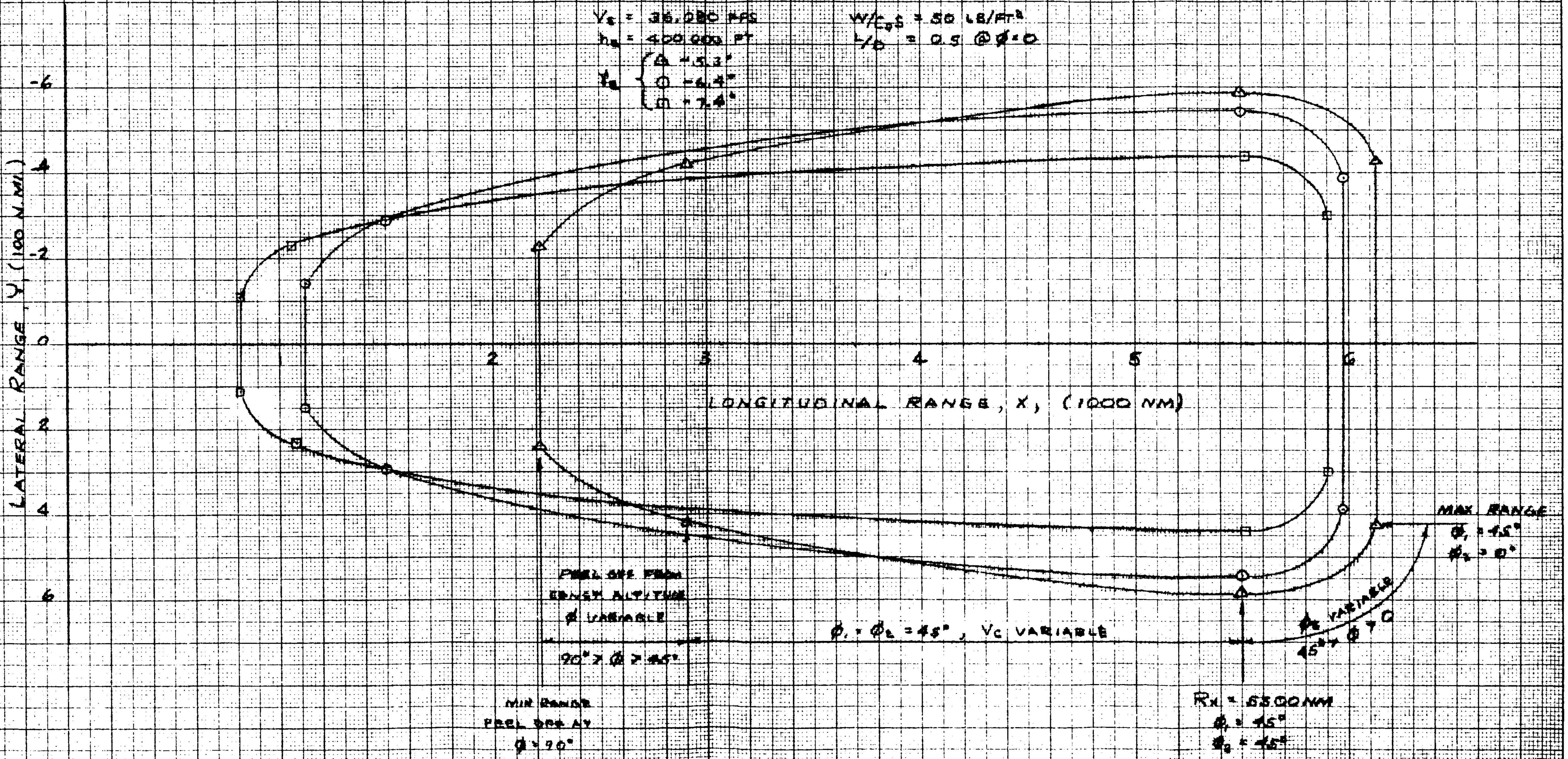
PREPARED BY:	NORTH AMERICAN AVIATION, INC. SPACE and INFORMATION SYSTEMS DIVISION	PAGE NO. 41, 42
CHECKED BY:		REPORT NO.
DATE:		MODEL NO. [redacted]

INERTIAL RANGE ENVELOPES

ENTRY CONDITIONS

V_E = 36,080 FPS
h_E = 400,000 FT
γ_E { △ -5.3°, ○ -6.4°, □ -7.4°

$W/C_D S$ = 50 LB/FT²
L/D = 0.5 @ ϕ = 0

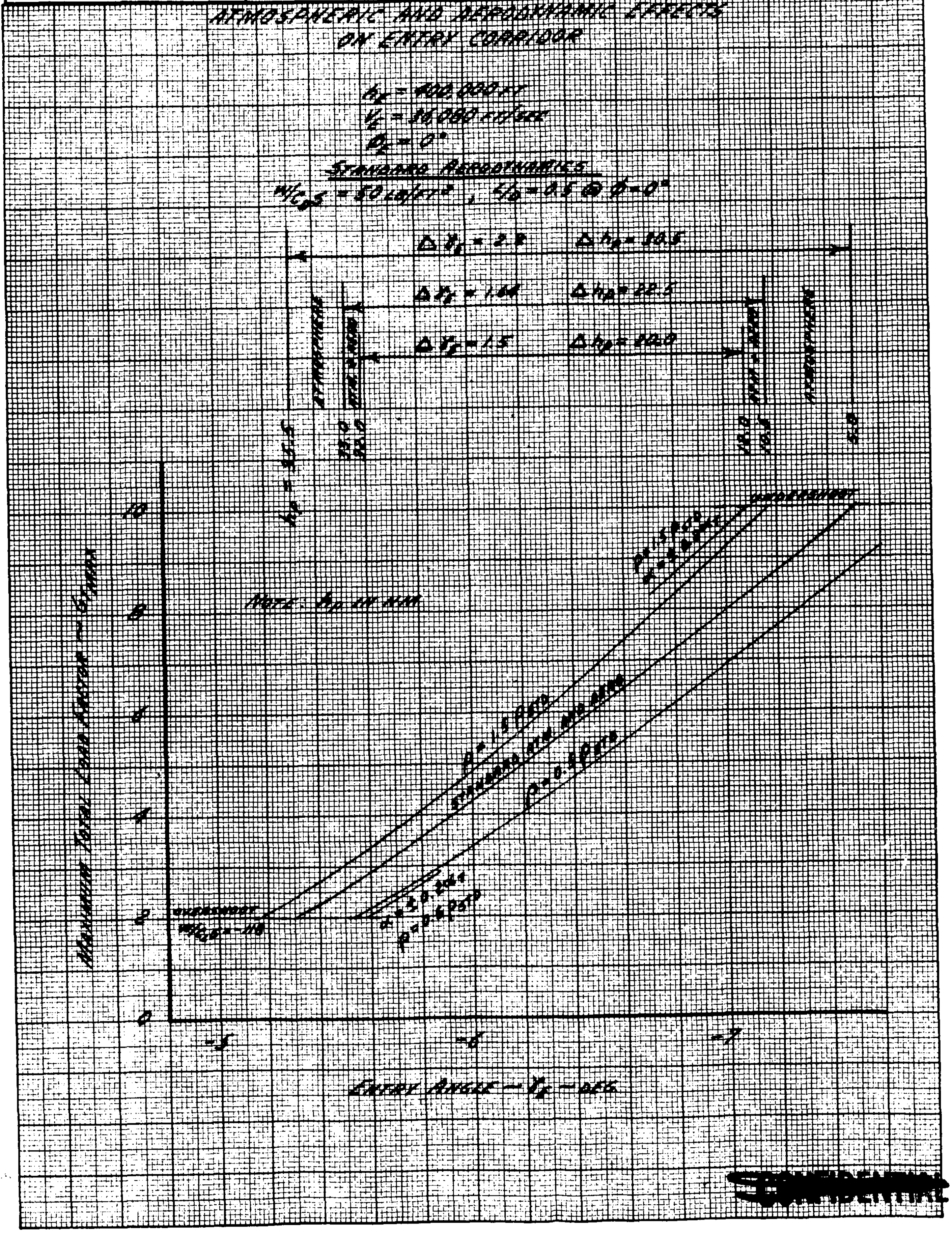


[redacted]

realized by selecting various velocities along the constant altitude path. The maximum velocity was limited to achieving 5500 nautical miles longitudinal range for each entry condition. For all entries the maximum range resulted in some amount of atmospheric exit. The minimum range was based upon rolling the vehicle to a 90 degree (L/D = 0) bank angle at satellitic speed from the constant-altitude path. For the undershoot entry this corresponds to approximately 8 g's. The entry conditions for these footprints correspond to the design corridor boundaries and a nominal entry angle. Evaluation of this flight mode as well as others will depend upon the range sensitivity to the trajectory parameters, guidance sensing capability, and range correction capability. Flight mode selection will be based upon these factors as well as emergency considerations during entry.

4.2.2.1.4.2.1 Corridor. - The characteristics of the atmospheric entry phase for lunar missions are dependent upon several entry corridors. The first, the design corridor, is obtained with a standard atmosphere and standard aerodynamics. Entry corridor characteristics are based upon the interface altitude of 400,000 feet and approximate escape speed. The overshoot and undershoot entry angles for this corridor, based upon entry at zero bank angle, are -5.3 degrees and -7.5 degrees respectively. These boundaries correspond to perigee altitudes of 35.5 and 5 nautical miles. The overshoot boundary is defined on the basis of an available $W/C_L S$ of -118 LB/FT^2 at the initial pull-out and corresponds to -0.85 C_L max. The total maximum load factor encountered along an overshoot entry is approximately 2. The undershoot boundary is based on a maximum total load factor of 10. Based upon these limits, the corridor depth is $\Delta \gamma EN$ = 2.2 degrees or Δh_p = 30.5 nautical miles. The vehicle systems are designed from the entry conditions of the design corridor. The second corridor is the operational corridor. It is obtained by considering atmospheric deviations, and aerodynamic deviations and systems tolerances which tend to decrease the corridor depth. Operation within this corridor would guarantee success with respect to loads and capture during the entry phase. Present estimates of this corridor depth are 20 nautical miles. The third corridor is the guidance corridor. As a design objective, this corridor should be approximately ±4 nautical miles from the desired entry condition. A guidance corridor of this depth would permit an advanced selection of the desired entry conditions and reduce touchdown dispersions for a specific range. Figure 4-10 illustrates the design and operational corridor load characteristics. The center curve and the upper arrowed line represent the design corridor. The decrease in corridor depth is indicated for an atmospheric density change of $\pm 0.5\rho_{STD}$ and trim angle-of-attack changes of $\pm 0.20\,\alpha_{TRIM}$. These data correspond to entry at a zero bank angle. The effect on corridor boundaries for entry at other than zero bank angle is illustrated in Figure 4-11. These data indicate that the corridor is not significantly changed for entry bank angles up to 30 degrees but is considerably narrowed as the entry bank angle approaches the ballistic value (ϕ = 90 degrees).

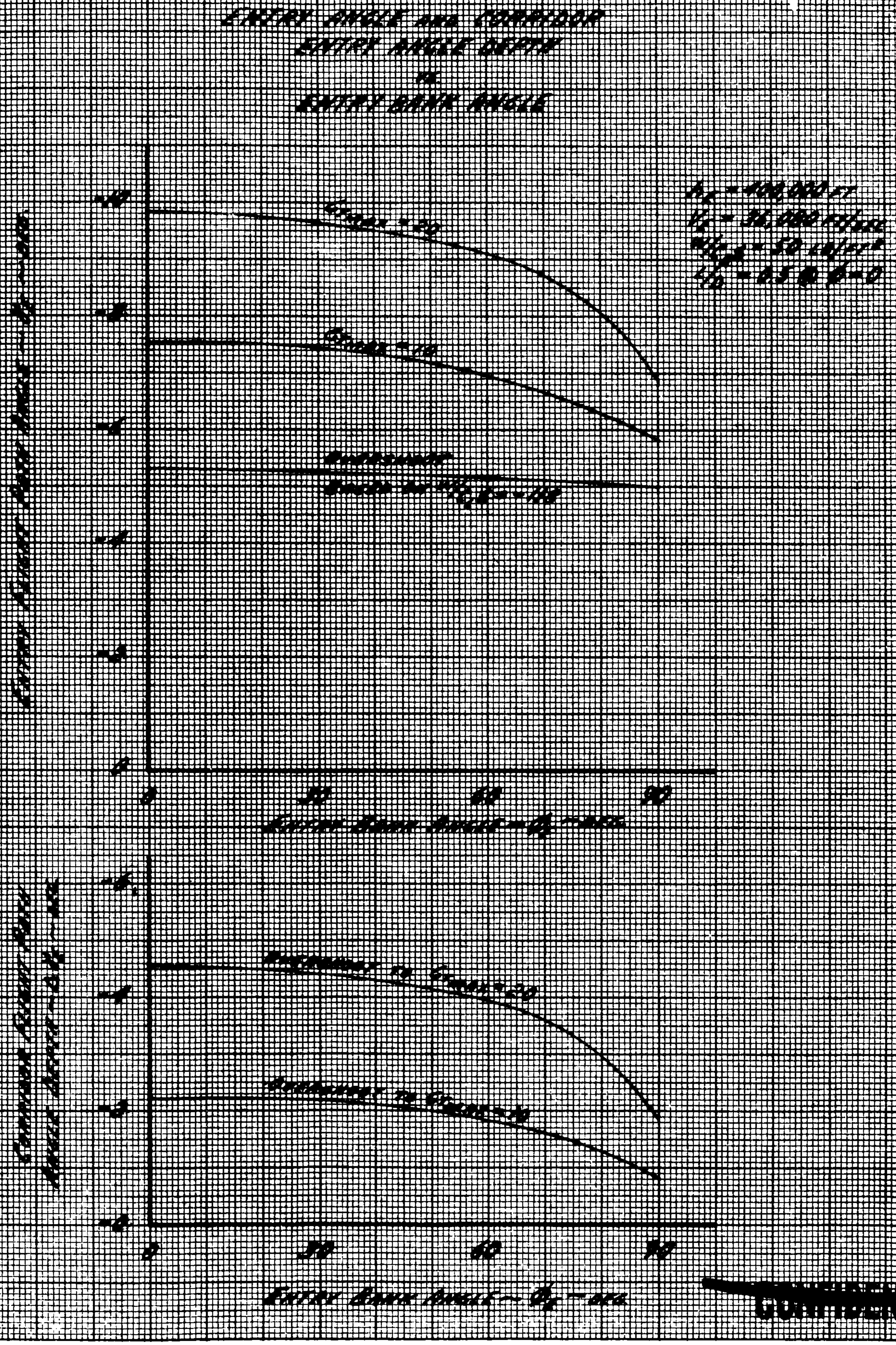
PREPARED BY:	NORTH AMERICAN AVIATION, INC. SPACE and INFORMATION SYSTEMS DIVISION Figure 4-10	PAGE NO. 44 OF
CHECKED BY:		REPORT NO. SID 62-338
DATE:	CORRIDOR CHARACTERISTICS	~~CONFIDENTIAL~~



~~CONFIDENTIAL~~

Figure 4-11

EFFECT OF BANK ANGLE ON CORRIDOR



~~CONFIDENTIAL~~

4.2.2.1.4.2.2 Emergency Mode. - The most important factor affecting the development of an emergency entry mode is that it should assure a high probability of crew survival. This implies not only that the entry maneuver be successfully executed, but that the landing should occur in a predictable area. The emergency mode should blend in well with the normal operating mode so that an immediate switch can be made at any point along the trajectory. A constant altitude mode whereby the pilot manually controls the bank angle such as to follow a predetermined constant altitude load-time history offers one possibility. The load-time history for this mode is presented in Figure 4-12 for the design corridor boundaries, a middle of the corridor nominal trajectory and two trajectories representing a ±5 nautical miles guidance corridor depth from the nominal. The load-time history prior to the peaks would be the same for all entries with a constant bank angle, but would vary subsequent to the peak depending on the particular mode. The constant altitude emergency mode is a logical companion to the skip-mode previously described since the load time histories are identical at least to the pull-up point. The dashed constant-velocity line indicates the start of a less sensitive region with respect to pilot responsibilities after which some amount of touchdown control could be initiated. Other modes and refinements to the present emergency mode will be investigated.

4.2.2.1.4.2.3 Recovery. - Same as 4.2.1.5.

4.2.3 Lunar Orbit Mission. -

4.2.3.1 Translunar. - The translunar phase of the lunar orbit mission must satisfy all of the requirements specified for the translunar portion of the circumlunar mission. Free return capability must exist in case injection into lunar orbit does not occur.

4.2.3.1.1 Initial Phase. -

4.2.3.1.1.1 Ascent. - Same as 4.2.2.1.1.

4.2.3.1.1.2 Orbit. - Same as 4.2.2.1.2.

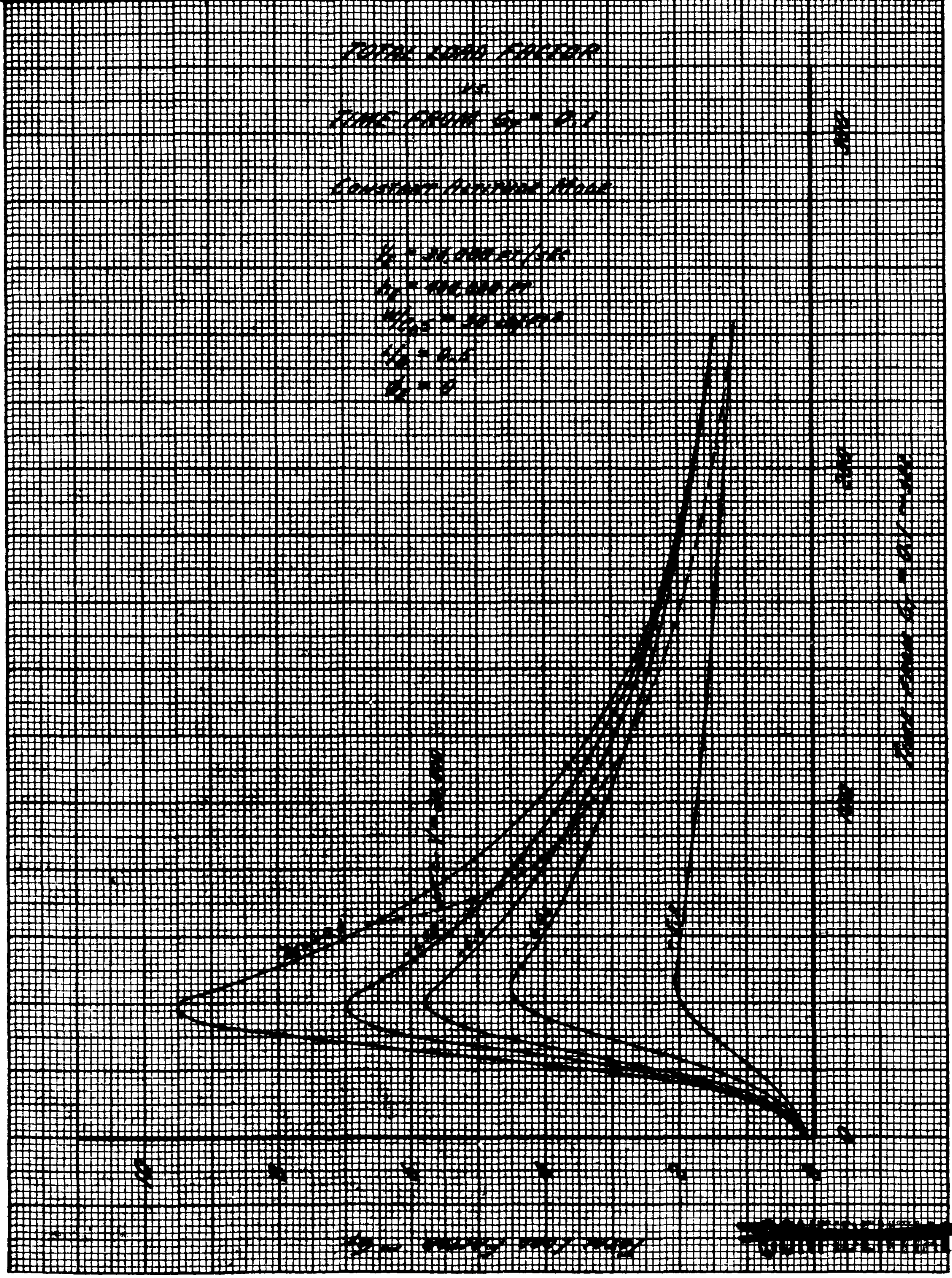
4.2.3.1.2 Mid-course Phase. -

4.2.3.1.2.1 Injection. - Same as 4.2.2.1.3.1.

4.2.3.1.2.2 Coast. - Same as 4.2.2.1.3.2.

4.2.3.1.3 Terminal Phase. -

PREPARED BY:	NORTH AMERICAN AVIATION, INC. MISSILE DIVISION	PAGE NO. 47 OF
CHECKED BY:	Figure A-12	REPORT NO. SID 62-338
DATE:	EMERGENCY MODE CHARACTERISTICS	MODEL ~~CONFIDENTIAL~~



4.2.3.1.3.1 Injection. - Injection into a circular lunar orbit having an altitude of 100 nautical miles will occur at perilune of the translunar trajectory. Figure 4-13 shows the perilune velocity as a function of transit time from translunar injection for several earth-moon distances. The transit time limits are established by trajectories whose transearth phase enter direct between inclinations of 0 and 62.5 degrees. The data in Figure 4-13 are based on trajectories having translunar plane inclinations of 5 degrees to the moon's orbit plane. Both the time to perilune and the velocity at perilune will increase with an increase in translunar inclination. Figure 4-13 also shows the velocity impulse required to inject into a circular orbit at perilune. The solid lines are simply the difference between perilune velocity and circular orbit velocity. The dashed lines include a 5 degree included angle between the velocity vectors. Injection and lunar orbit geometry is illustrated in Figure 4-14. A performance capability will be provided to transfer from circular to elliptical lunar orbits.

4.2.3.1.3.2 Orbit. - Injection into a 100 nautical mile circular lunar orbit from a free return circumlunar trajectory restricts the circular orbit inclination to the moon's orbit plane to within about ±10 degrees. A 5 degree maneuver capability at injection increases these limits to ±15 degrees. Since the inclination of the moon's orbit plane to the lunar equator plane is 6.68 degrees (constant), the resulting inclination limits between the circular orbit plane and the lunar equator plane are about 0 to 22 degrees. The circular orbit's ascending node on the lunar orbit plane can assume any orientation from 0 to 360 degrees at orbit injection depending on the particular combination of translunar and transearth inclinations. The orientation of the node with respect to the earth-moon line of centers will change with time elapsed in orbit, however, by the angular rotational rate of the earth-moon system. The limit on time elapsed while orbiting the moon is established by the difference between the maximum total mission time (14 days) and the sum of the translunar and transearth flight times. This is approximately eight days. Consideration of daylight landing at earth would reduce this to four days.

A lunar satellite orbit is affected by perturbations from many sources, primarily however, by the attractions of the lunar triaxial ellipsoid and by the planet Earth. Of these two, the lunar triaxiality predominates. For instance, for a circular orbit at 100 nautical miles altitude it exceeds the effects of the earth by a factor of 14 to 1. The effects of the sun and the planets are negligible. Approximate analytical, as well as numerical, investigations in this area are being conducted to determine the combined perturbational effects of earth and asymmetrical moon as functions of orbital, positional, and nodal parameters. Although the various approaches have not yet yielded a comprehensive overall theory, preliminary results do indicate some representative values for the regression of the node and for altitude oscillations in typical orbits.

Figure 4-13

LUNAR ORBIT INJECTION REQUIREMENTS

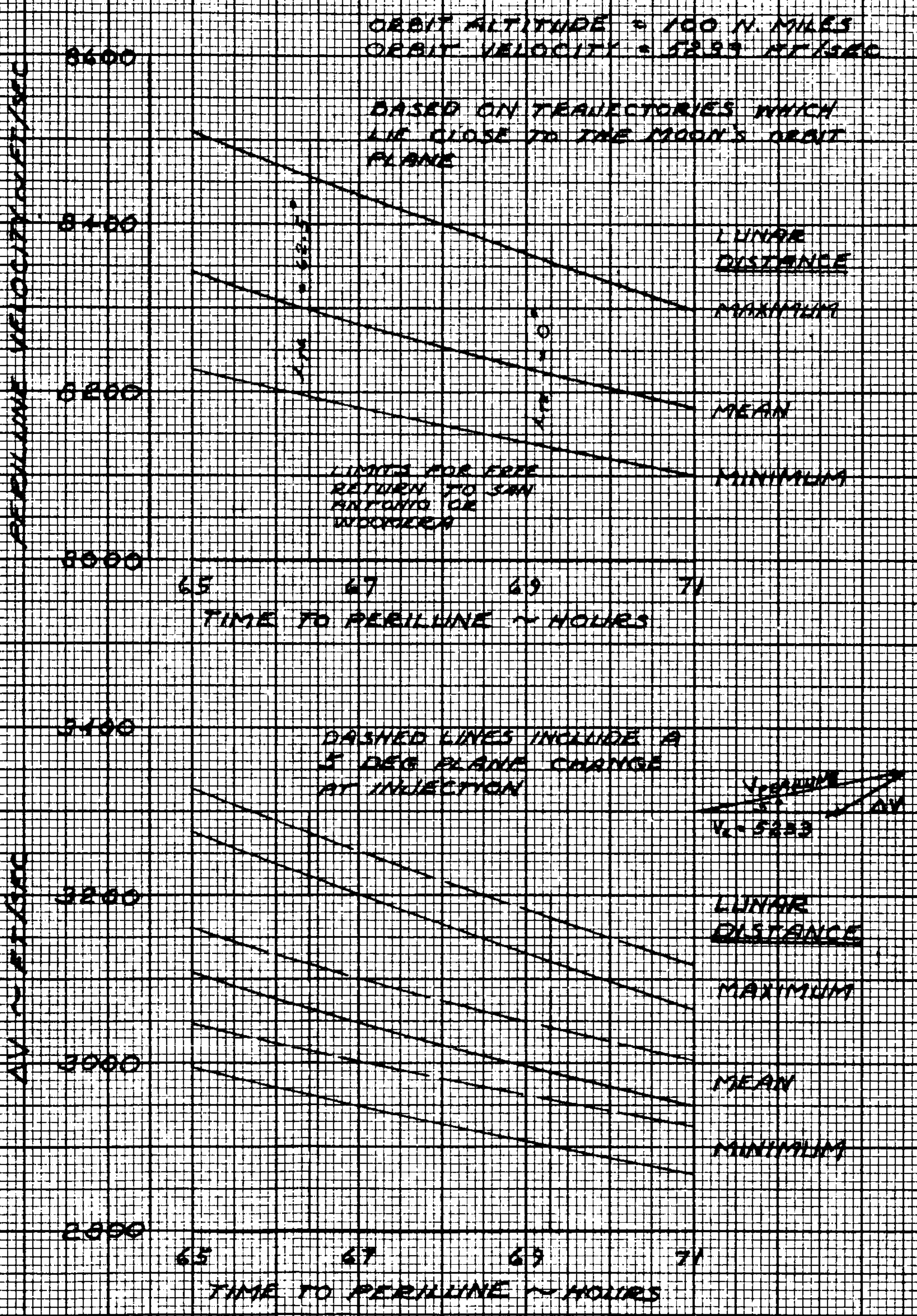
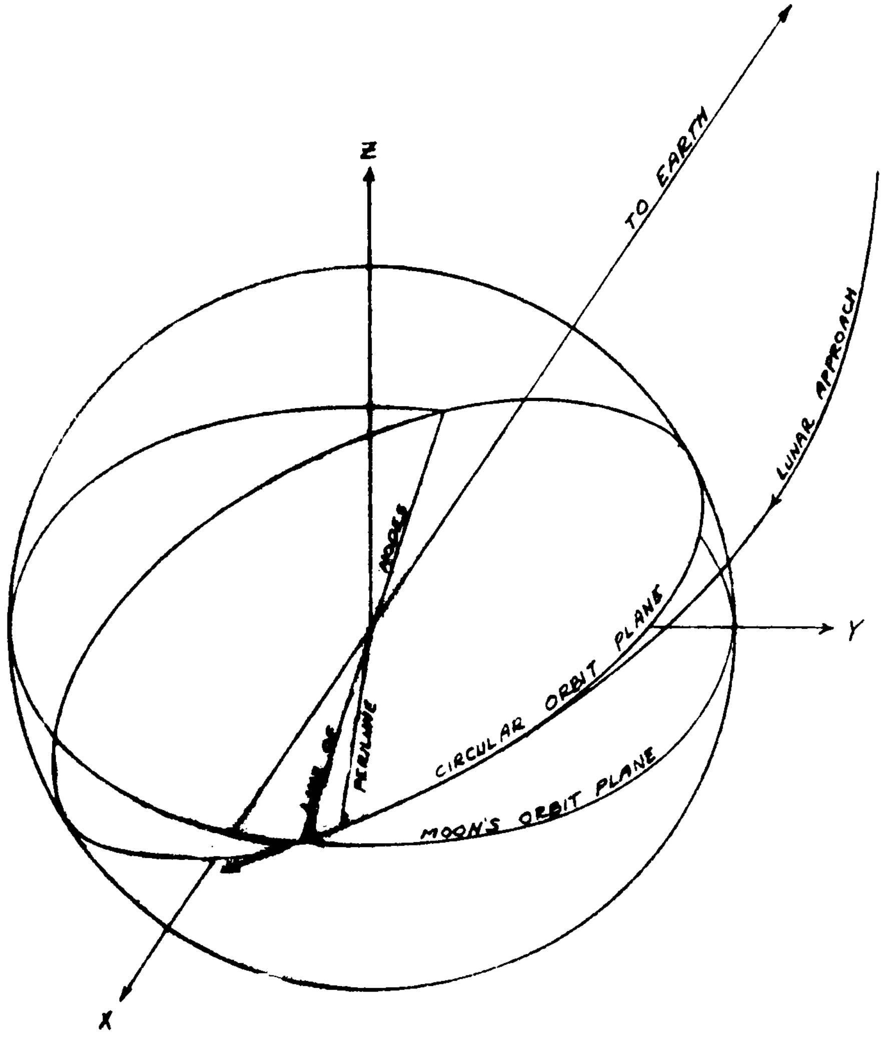


Figure 4-14

LUNAR ORBIT INJECTION GEOMETRY


Z
TO EARTH
LUNAR APPROACH
NODES
LINE OF
PERILUNE
CIRCULAR ORBIT PLANE
Y
MOON'S ORBIT PLANE
X

Specifically, for a nominally circular orbit at 100 nautical miles altitude over the average lunar surface at an inclination of 10 degrees to the lunar equator, analytical results indicate a secular regression of the node of 1.15 degrees per day. Figure 4-15 presents the dependence of this rate upon orbital inclination. Numerical integration techniques have yielded a similar value. It should be noted that for retrograde orbits, of interest to the Apollo mission, the motion of the node will actually be positive (counterclockwise), rather than negative. Similarly, for the radial oscillations of the vehicle path, as presented on Figure 4-16, a representative total amplitude of 3, 500 feet has been numerically computed for the above typical orbit, under neglect of the earth's attraction. The latter has been found, again by numerical integration, to be contributing an additional oscillation in altitude of the order of 250 feet. The inclination of the typical orbit of 10 degrees has reduced the minimum altitude by about 43 feet from the equatorial orbit value. Minimum altitude variation near-circular orbits may be achieved by increased injection velocity in order to reduce the altitude oscillations with respect to the nominal surface; values of less than 250 feet amplitude may be achieved.

It may, therefore, be concluded that stable lunar orbits may be achieved at the 100 nautical miles altitude despite the formidable perturbing effects of Earth and triaxial Moon. The predicted positional and out-of-plane deviations accumulated in a perturbed lunar orbit must be considered in the determination of velocity increment and direction required for injection into the transearth return trajectory.

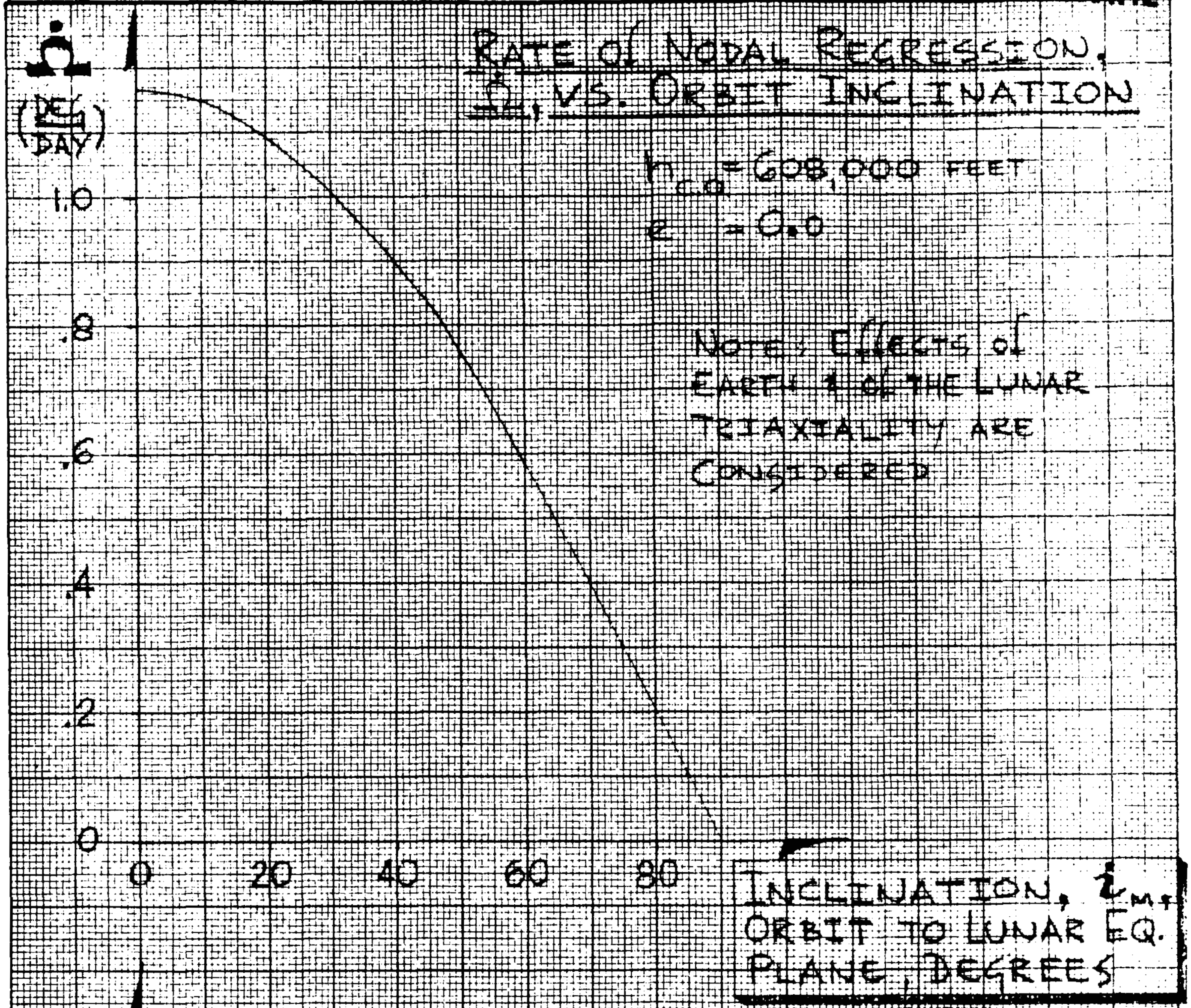
4.2.3.2 Transearth. -

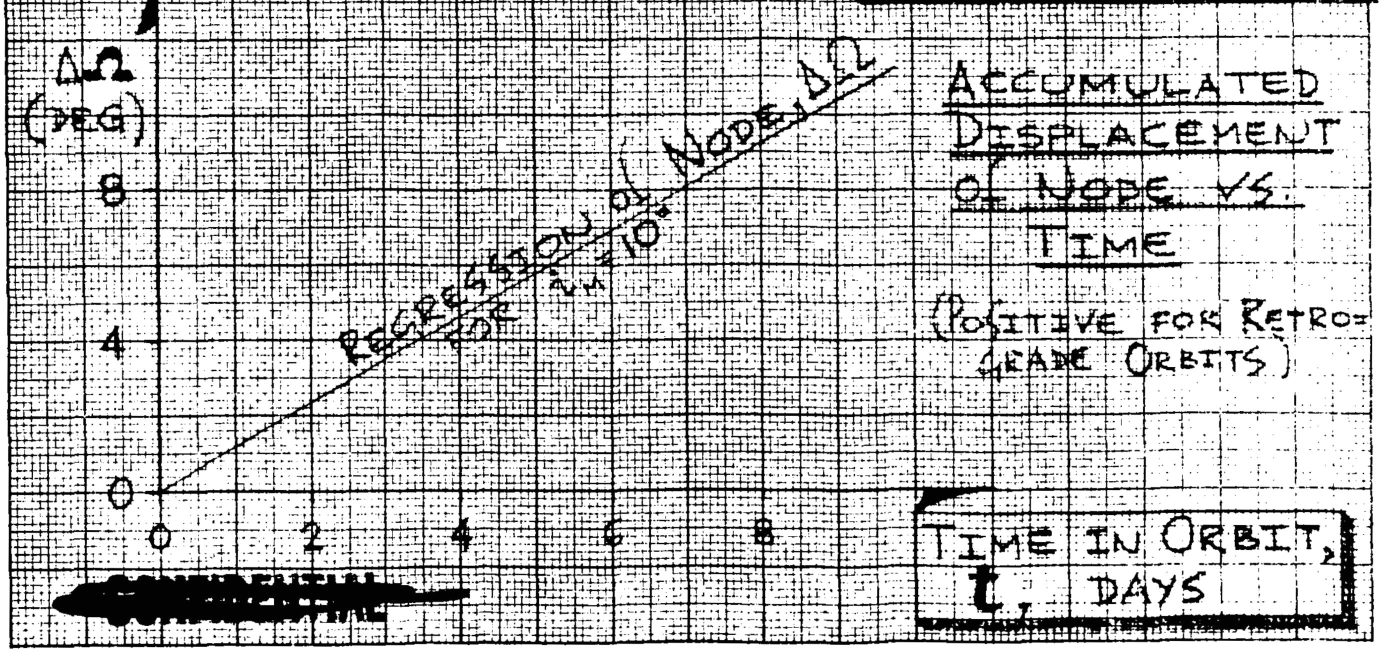
4.2.3.2.1 Initial Phase. - The circular lunar orbit is effectively a parking orbit for injection into the transearth phase.

4.2.3.2.2 Mid-Course Phase. -

4.2.3.2.2.1 Injection. - Injection into the trans-earth trajectory must be possible at least once during each revolution in the circular orbit. The position in the orbit, velocity magnitude, and out-of-plane velocity component at trans-earth injection control the flight path angle, inclination, and time at earth entry. Figure 4-17 illustrates the manner in which the ΔV required for injection varies as a function of trans-earth plane inclination and the circular orbit ascending node position on the moon's orbit plane (measured CCW from the earth's direction at the time of injection). The data are for an injection velocity of 8, 494 ft/sec and a circular orbit plane inclination of 15 degrees. The minimum ΔV for each inclination curve occurs when the out-of-plane component is zero and the magnitude is the difference between injection velocity and circular orbit velocity. The maximum ΔV occurs when the out-of-plane component is greatest. Figure 4-18 shows the maximum ΔV and out-of-plane angle required for injection as a function of transearth

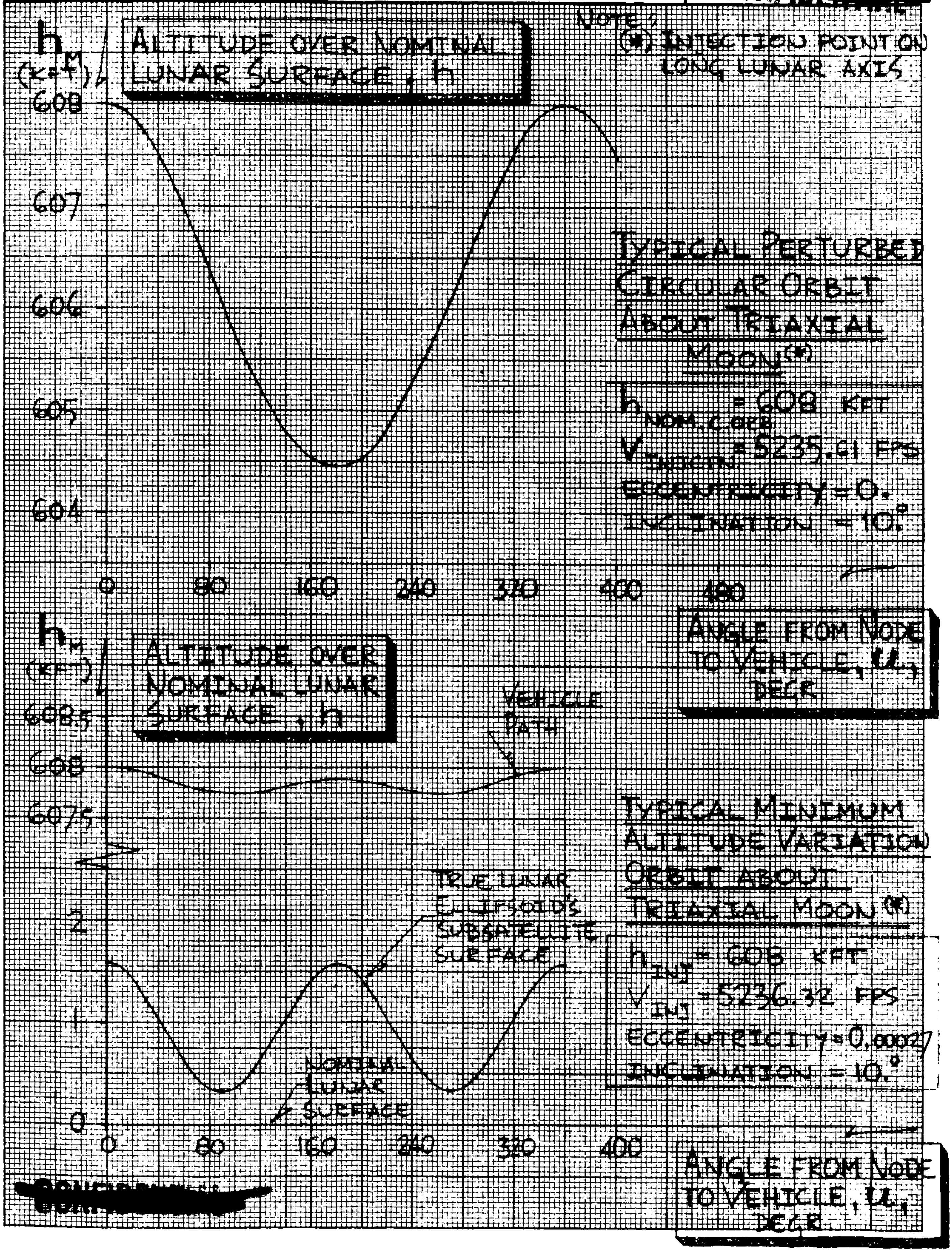
Figure 4-15


RATE OF NODAL REGRESSION, $\dot{\Omega}$, VS. ORBIT INCLINATION


ACCUMULATED DISPLACEMENT OF NODE VS. TIME

(POSITIVE FOR RETROGRADE ORBITS)

PREPARED BY: ABS	NORTH AMERICAN AVIATION, INC. SPACE and INFORMATION SYSTEMS DIVISION Figure 4-16	PAGE NO. 53 OF
CHECKED BY:		REPORT NO. SID 62-338
DATE: FEB 14, 62.	LUNAR SATELLITE PERTURBATIONS	MODEL [redacted]

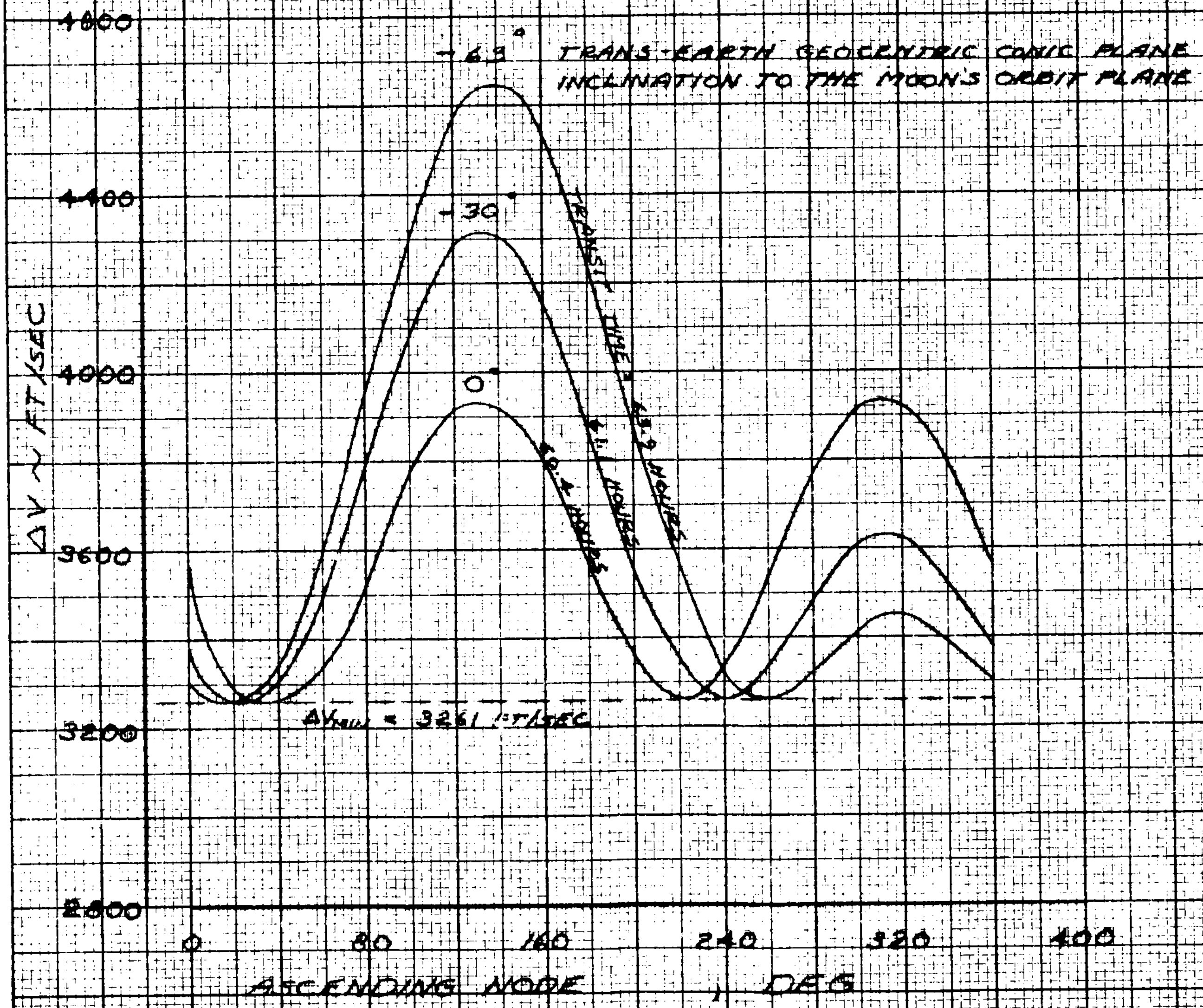


[redacted]

Figure 4-17

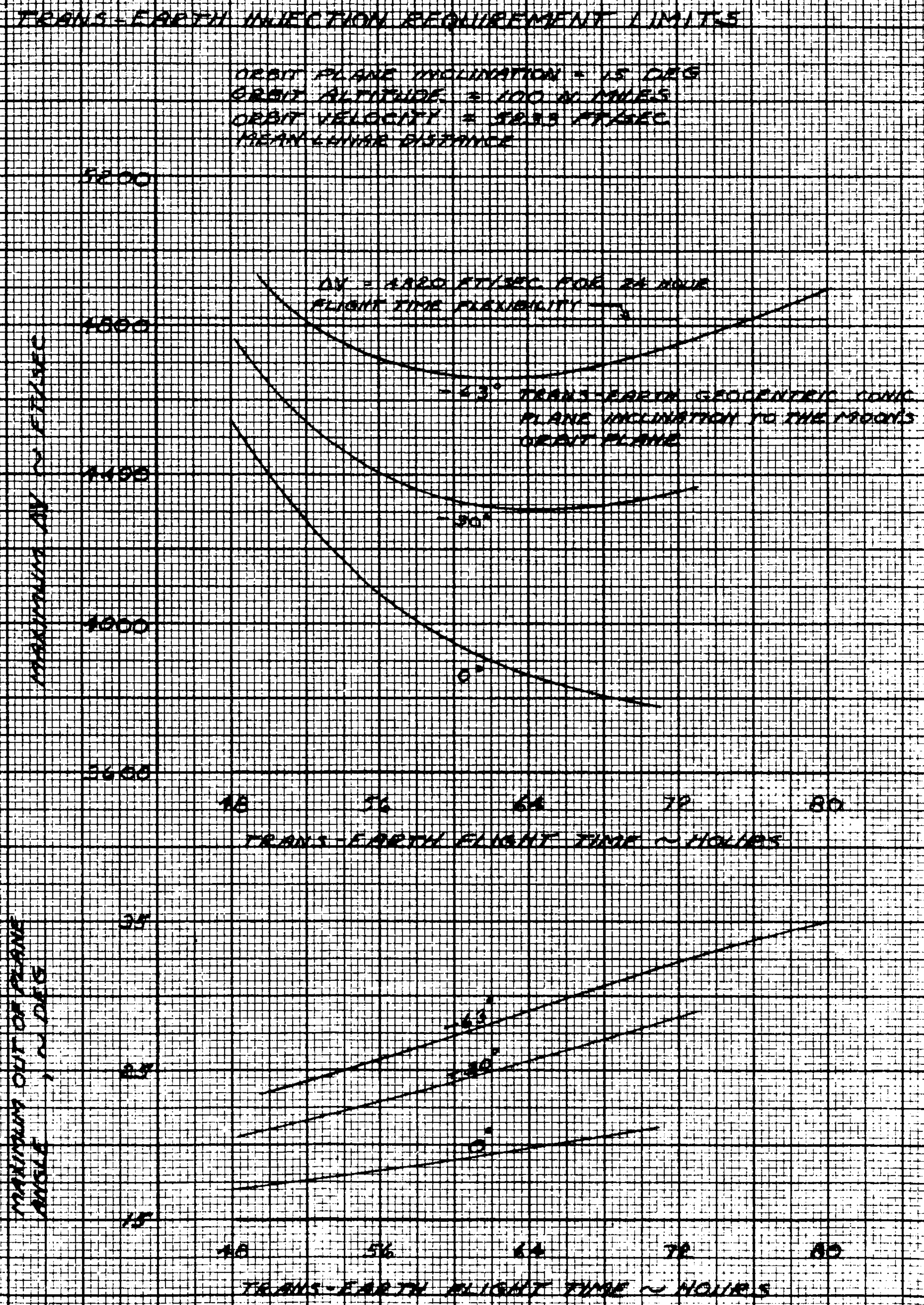
VELOCITY INCREMENT REQUIRED FOR TRANS-EARTH INJECTION

ORBIT PLANE INCLINATION = 15 DEG
ORBIT ALTITUDE = 100 N. MILES
ORBIT VELOCITY = 5233 FT/SEC
INJECTION VELOCITY = 8494 FT/SEC
MEAN LUNAR DISTANCE



[redacted]

Figure 4-18



[redacted]

flight time. The horizontal dashed line indicates that for a 24-hour flight-time flexibility, required to return to a specific landing site, the ΔV necessary for injection will not exceed 4,820 ft/sec regardless of the orientation of the circular orbit plane line of nodes.

4.2.3.2.2.2 Coast. - The transearth trajectory must satisfy the same entry conditions established for the transearth portion of the circumlunar trajectory. The trajectory will lie below the moon's orbit plane for returns to San Antonio and above the moon's orbit plane for returns to Woomera.

4.2.3.2.3 Terminal Phase. - 4.2.2.1.4.

4.2.4 Lunar Landing Missions (Phase C). -

4.2.4.1 Earth Orbital Rendezvous. - No firm requirements for this mission have been established.

4.2.4.2 Lunar Orbital Rendezvous. - No firm requirements for this mission have been established.

4.2.4.3 Direct. -

4.2.4.3.1 Translunar. -

4.2.4.3.1.1 Initial Phase. -

4.2.4.3.1.1.1 Ascent. - The Apollo vehicle is launched from the AMR using the Nova.

4.2.4.3.1.1.2 Orbit. - An earth parking orbit will be used to provide launch time flexibility for the direct launch mode. For the direct mode the parking orbit will be 100 nautical miles.

4.2.4.3.1.2 Mid-course Phase. -

4.2.4.3.1.2.1 Injection. - The powered trajectory phase from earth orbit to translunar injection is performed by the third stage of the Nova Vehicle.

4.2.4.3.1.2.2 Coast. - The translunar trajectory is designed to provide a "free-return" circumlunar trajectory should an abort be necessary prior to perilune. Midcourse corrections during the translunar coast period are made using the lunar landing module landing engine. The maximum midcourse correction capability shall be 500 ft/sec.

4.2.4.3.1.3 Terminal Phase. -

4.2.4.3.1.3.1 Injection. - Figure 4-19 shows the spacecraft trajectory in the lunar vicinity. At perilune of the translunar trajectory the velocity increment required for injection into a 100 nautical mile orbit, including a 5 degrees planar change, will be provided by the retrograde engines. Velocity vector control and vernier velocity control will be provided by the landing engine.

4.2.4.3.1.3.2 Orbit. - The spacecraft will make one pass over the landing site prior to initiating the landing maneuver at 100 nautical miles.

4.2.4.3.1.3.3 Descent. -

4.2.4.3.1.3.3.1 Lunar Orbital Transfer. - A Hohmann transfer from 100 nautical miles to 50,000 feet will be initiated by reducing the vehicles orbital velocity by approximately 115 ft/sec using the landing module's landing engine.

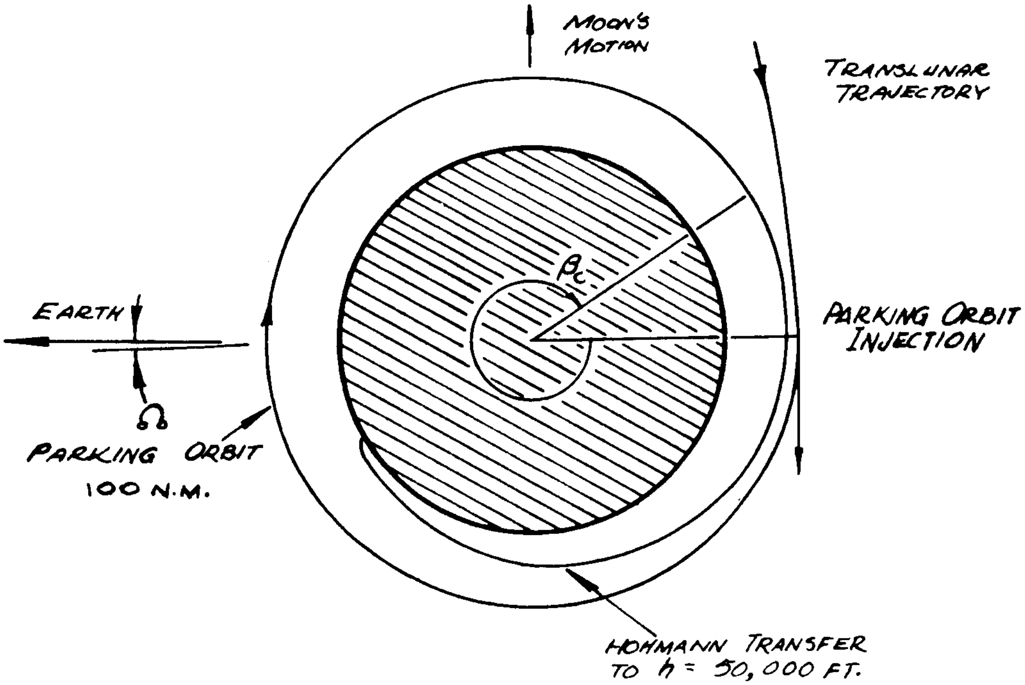
4.2.4.3.1.3.3.2 Main Retro. - The main landing maneuver is initiated at the end of the Hohmann transfer. This maneuver reduces the vehicle's velocity and altitude to small values prior to the landing phase. Two alternate steering modes are being considered for the main retro maneuver. The two modes differ in their direction of approach to the terminal point. Figure 4-20 shows the required main retro characteristic velocity for a vertical approach to the terminal point, as a function of the initial thrust to weight ratio, while Figure 4-21 corresponds to a horizontal approach.

4.2.4.3.1.3.3.3 Landing. - The landing maneuver consists of a hover, translation, and landing mode beginning at an end retro altitude of 1,000 ft. The hover and translation requirements for the terminal maneuver have not, as yet, been specified. For preliminary design purposes 1,000 ft/sec characteristic velocity has been added to the total requirements for the lunar landing mission to account for the landing maneuver as well as propulsion anomalies and control losses.

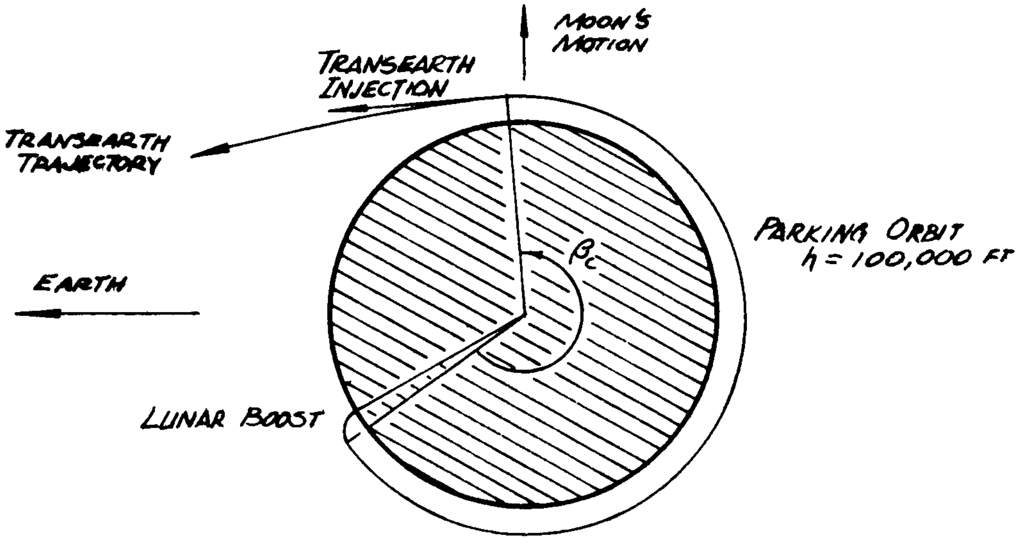
4.2.4.3.2 Transearth. - The transearth phase begins at lunar launch and ends at earth recovery. All propulsion during this phase is provided by the service module. The total characteristic velocity required to perform the various mission segments is shown in Figure 4-22 as a function of the launch thrust to weight ratio. The transearth trajectory requirements are based on impulsive vertical boosts from the surface of a spherical moon which is in a circular orbit about a spherical earth. The simplified model is the most versatile for obtaining a first approximation of the requirements at injection into the transearth trajectory for both the lunar orbit and lunar

Figure 4-19

LUNAR LANDING MISSION GEOMETRY


MOON'S MOTION
TRANSLUNAR TRAJECTORY
β_c
EARTH
PARKING ORBIT INJECTION
Ω
PARKING ORBIT 100 N.M.
HOHMANN TRANSFER TO h = 50,000 FT.

TRANSLUNAR


MOON'S MOTION
TRANSEARTH INJECTION
TRANSEARTH TRAJECTORY
β_c
PARKING ORBIT h = 100,000 FT
EARTH
LUNAR BOOST

TRANSEARTH

[redacted]

PREPARED BY:

CHECKED BY:

DATE: 11 FEB. 196

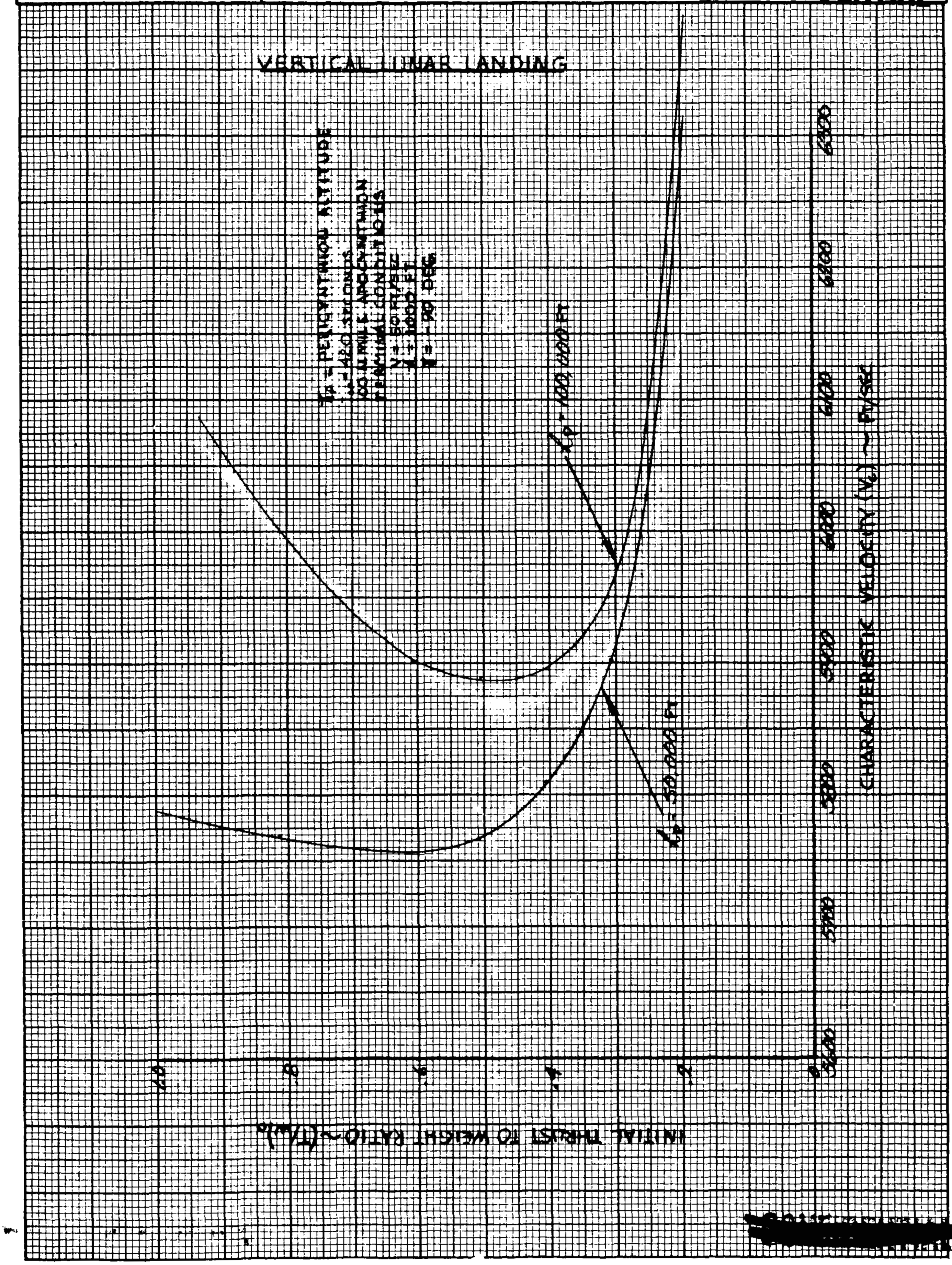
NORTH AMERICAN AVIATION, INC.
MISSILE DIVISION
Figure 4-20

LANDING MODULE PROPULSION REQUIREMENTS

PAGE NO. 59 OF

REPORT NO. SID 62-338

MODEL [redacted]



[redacted]

PREPARED BY: pwh

CHECKED BY:

DATE: 10 FEB. 1962

NORTH AMERICAN AVIATION, INC.
MISSILE DIVISION

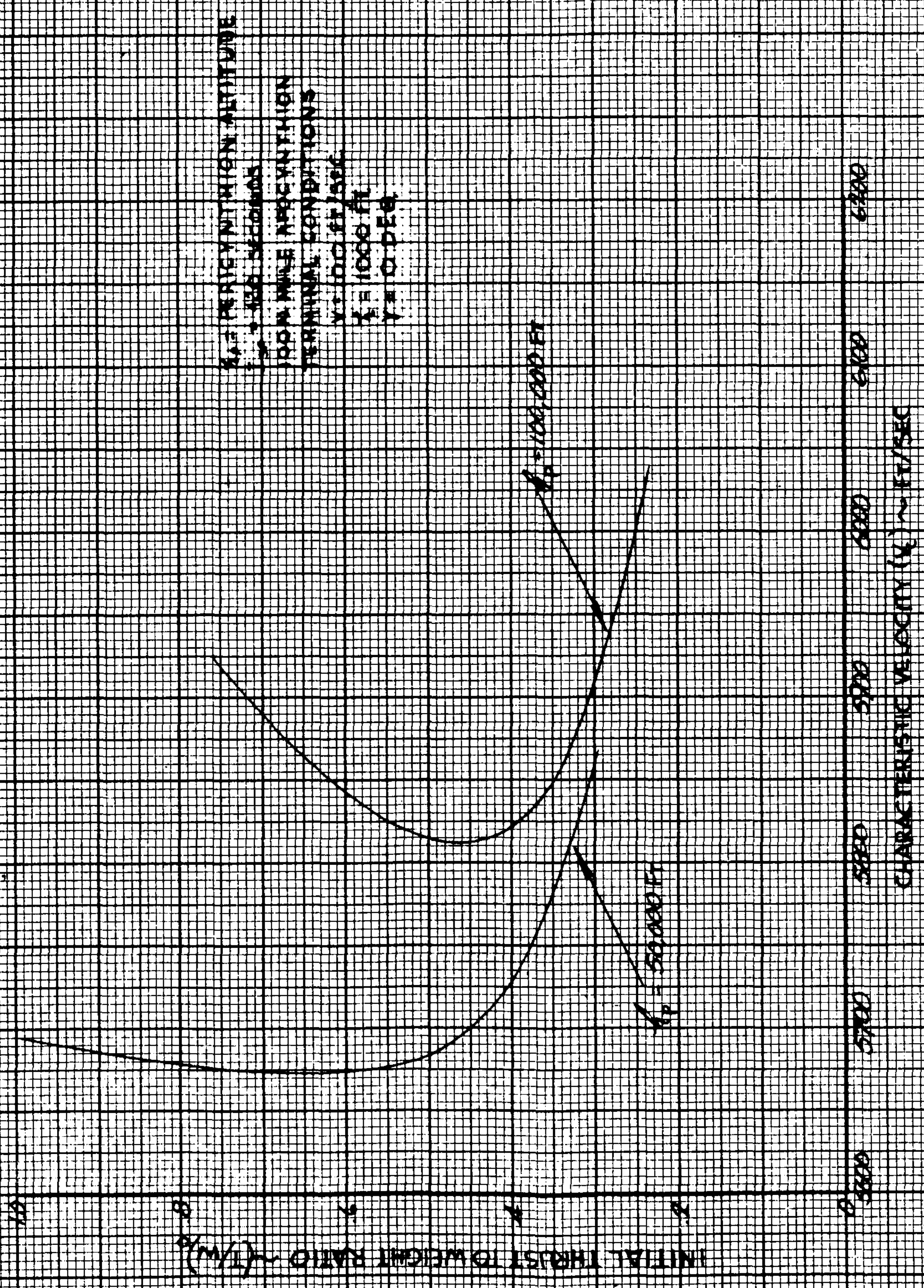
Figure 4-21

LANDING MODULE PROPULSION REQUIREMENTS

REPORT NO. SID 62-338

MODEL NO. ~~CONFIDENTIAL~~

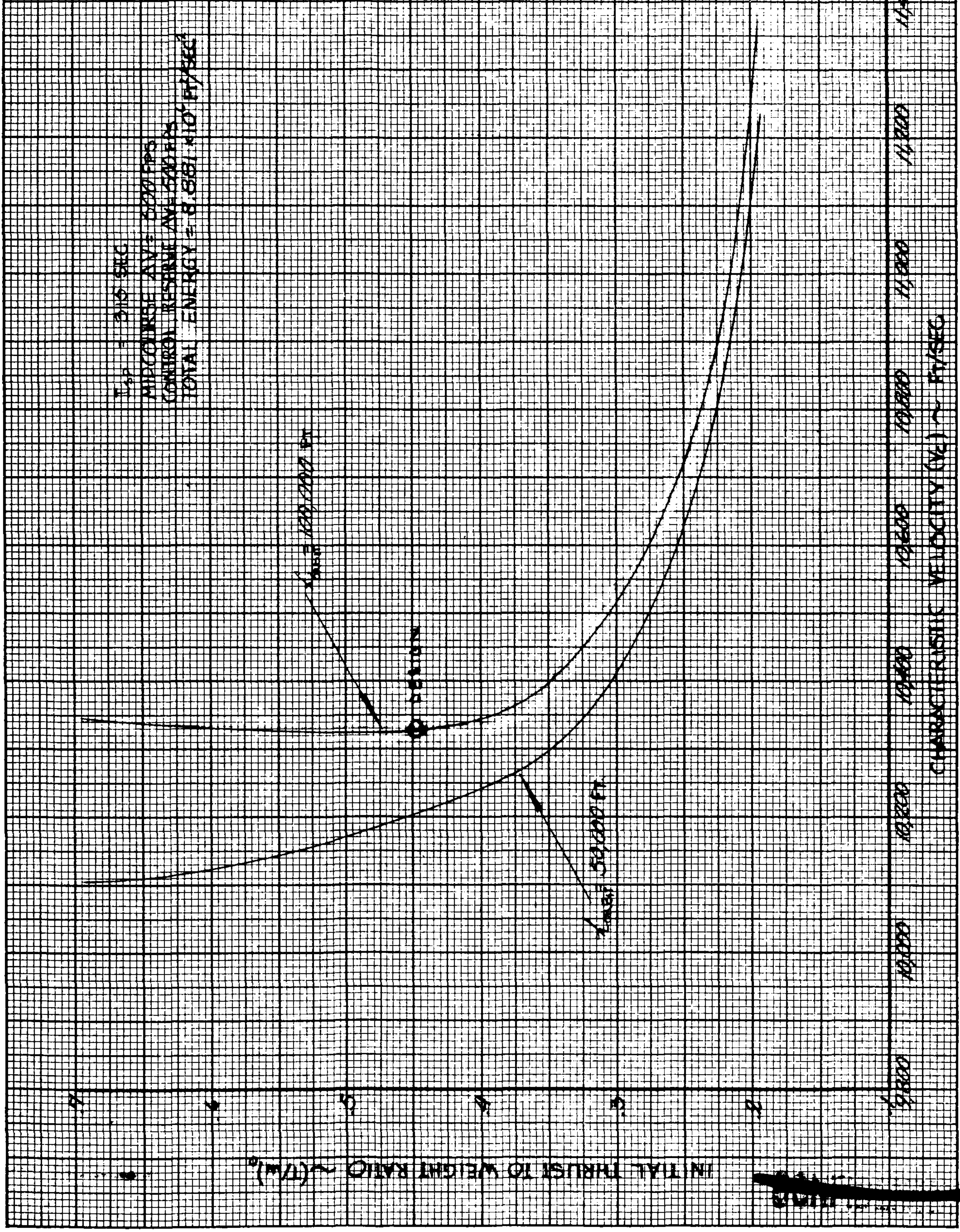
HORIZONTAL LUNAR LANDING



~~CONFIDENTIAL~~

PREPARED BY:	**NORTH AMERICAN AVIATION, INC.** **MISSILE DIVISION**	PAGE NO. 61 OF
CHECKED BY:		REPORT NO. SID 62-338
DATE: 12 FEB. 1962	Figure 4-22	MODEL: [redacted]

SERVICE MODULE
PROPULSION REQUIREMENTS



[redacted]

landing missions. The transearth trajectory problem can thereby be studied independently of the orientation of the lunar parking orbit or the location of the lunar landing site. Propulsion requirements are established by transearth injection out of a 100, 000 foot lunar orbit.

4.2.4.3.2.1 Initial Phase. - The initial phase of the transearth trajectory consists of boosting the vehicle into a 100, 000 ft. circular parking orbit. The vehicle remains in the parking orbit until it is in the proper position for injection into the transearth trajectory. Multiple lunar orbits prior to earth return may be used for additional lunar surveillance.

4.2.4.3.2.1.1 Ascent. - The thrust of the service module engine is 20, 000 pounds and the resulting launch thrust to weight ratio, which is .447, minimizes the required total characteristic velocity. The spacecraft is launched vertically from the lunar surface using the lunar landing module as a launch platform and is powered continuously into a 100, 000 foot circular lunar orbit.

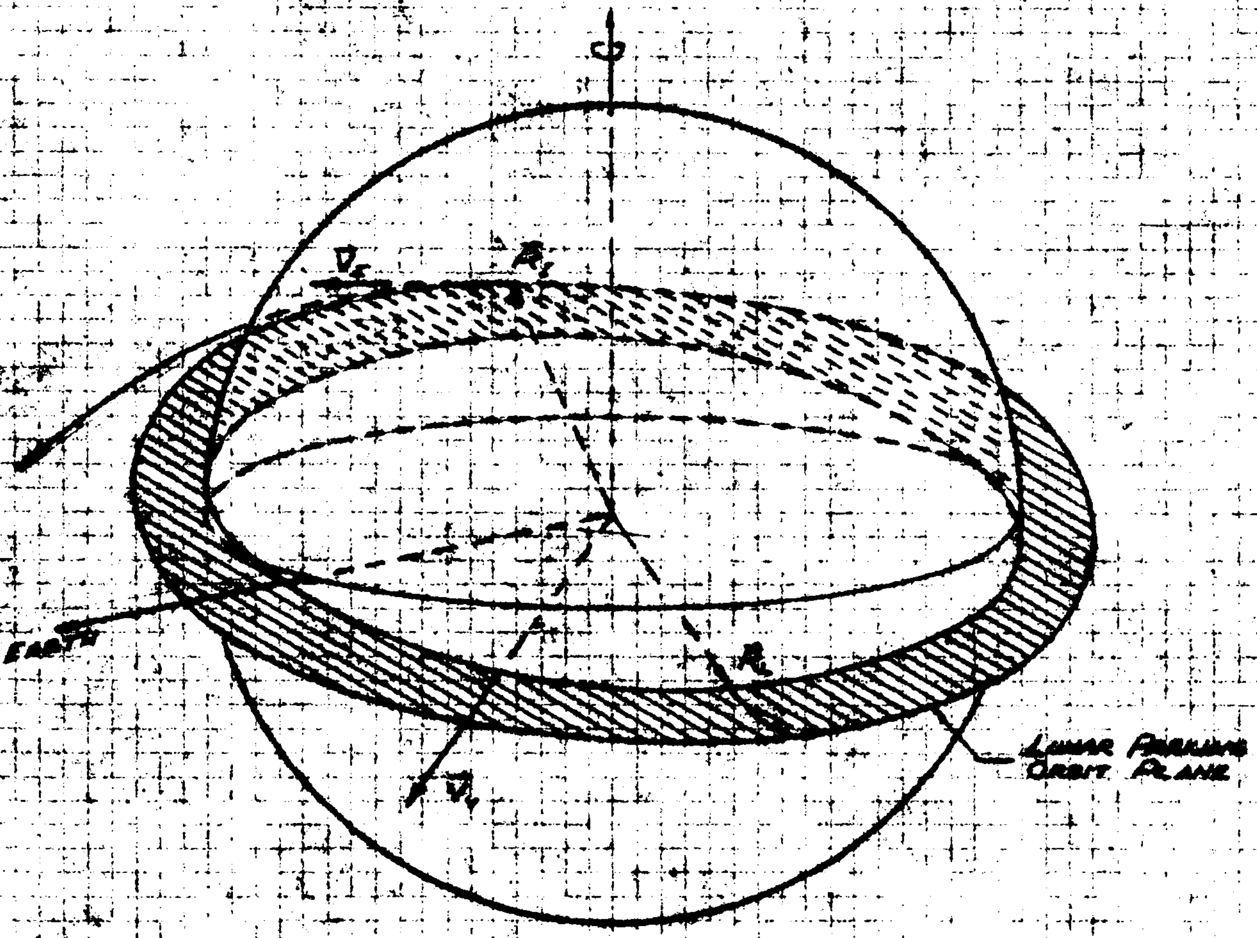
To a first approximation, any vehicle which injects into a transearth trajectory and satisfies a given set of terminal conditions must leave the lunar vicinity along one of a family of selenocentric conics which have the same energy and parallel asymptotes. The error in the approximation is due to the change in the potential field as the plane of the conic is varied. Therefore, the vertical boost analysis can be used to make a first approximation of the required energy and asymptote direction at injection to satisfy a given set of terminal conditions. For a given launch time, three terminal conditions must be satisfied; the inclination of the geocentric conic relative to the earth's equatorial plane, the entry flight path angle, and the entry time. The three terminal restraints must be satisfied by varying the conditions at injection into the transearth trajectory. The plane of the lunar parking orbit, and thus the launch azimuth, is defined by the lunar landing site radius vector and the vertical boost velocity vector which satisfies the required terminal conditions as shown in Figure 4-23.

4.2.4.3.2.1.2 Orbit. - The direction of the asymptote of the selenocentric conic is defined by the total impulse applied and the position of the vehicle in the parking orbit at the time of injection into the transearth trajectory. The required position at injection is determined by requiring the asymptote of the selenocentric conic to be parallel to the vertical boost velocity vector as shown in Figure 4-23.

4.2.4.3.2.2 Mid-course Phase. - The mid-course phase begins with the injection into the transearth trajectory and terminates at the entry altitude of 400, 000 ft. It includes the transearth coast phase with its associated mid-course corrections.

Figure 4-23

TRANSEARTH INJECTION GEOMETRY


V_2
R_2
EARTH
R_1
V_1
LUNAR PARKING ORBIT PLANE

V_1 ~ VERTICAL BOOST VELOCITY VECTOR
R_1 ~ LANDING/LAUNCH SITE RADIUS VECTOR
V_2 ~ TRANSEARTH INJECTION VELOCITY VECTOR
R_2 ~ TRANSEARTH INJECTION RADIUS VECTOR

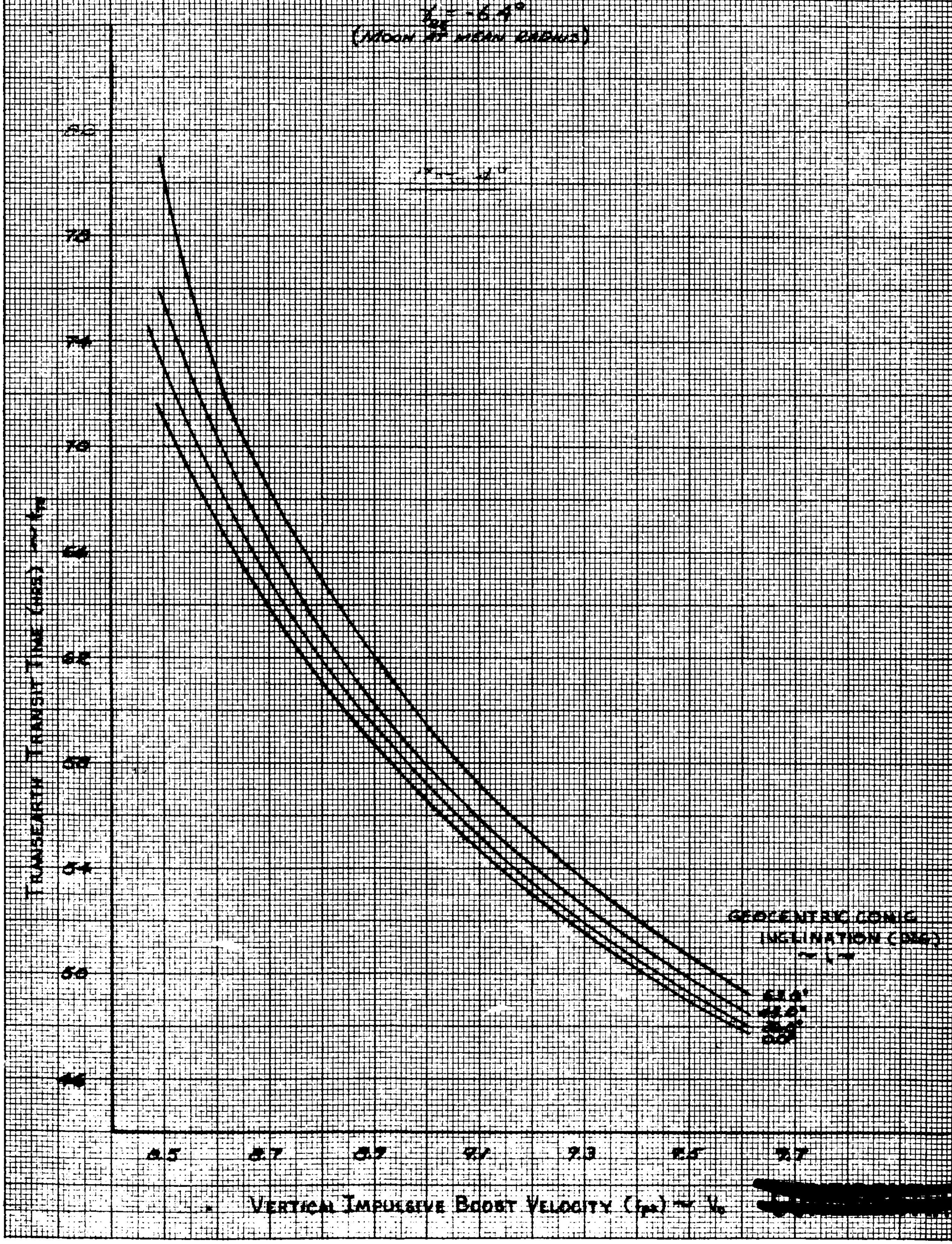
~~CONFIDENTIAL~~

4.2.4.3.2.2.1 Injection. - The total impulse at injection into the transearth trajectory must be such that the energy of the resultant selenocentric conic is equal to the energy of the vertical boost rectilinear conic. The required impulse is obtained by relighting the service module engine. Any set of terminal conditions can be satisfied if sufficient total impulse is available and there are no lunar launch azimuth restrictions.

Figures 4-24, 4-25, and 4-26 show some of the results of the vertical boost analysis. Figure 4-24 shows the transearth transit time as a function of the vertical boost velocity for an entry flight path angle of -6.4 degrees. It can be seen that the transit time can be varied by as much as twenty-four hours if sufficient total impulse is provided. Therefore, the vehicle could launch from the lunar surface at any time and return to a single earth landing site.

It should be noted that the transit time increases as the inclination of the earth conic relative to the plane of the moon's orbit about the earth increases. The lunar orbit mission requires inclinations between 29.48 and 33.99 degrees during the terminal phase of the mission. The same inclination requirements have been established for the circumlunar, lunar orbit, and lunar landing missions to minimize the GOSS requirements during entry. The inclination of the earth conic relative to the plane of the moon's orbit varies throughout the month if the inclination of the conic relative to the equatorial plane is held constant. To stay within the required inclination band with respect to the equatorial plane, the inclination relative to the moon's orbit plane must vary between 0.885 (29.480 - 28.595) and 62.55 (28.595 + 33.990) degrees when the inclination of the moon's orbit is a maximum (28.595 degrees). When the inclination of the moon's orbit is a minimum (18.305 degrees), the inclination of the earth conic must vary between 11.165 and 52.295 degrees. Figure 4-24 shows the variation in transit times over the required inclination range. Figure 4-25 shows the inclination of the geocentric conic (ι) relative to the moon's orbit plane as a function of the right ascension (θ) and declination (Φ) of the vertical boost launch site for an entry angle of -6.4 degrees. Figure 4-26 shows the vertical boost velocity as a function of launch site right ascension (θ) and declination (Φ) and the inclination of the geocentric conic (ι). The right ascension (θ) is measured in the plane of the moon's orbit counterclockwise from the earth-moon line of centers at the time of launch. The declination (Φ) is measured normal to the moon's orbit plane. It should be noted that the model is symmetrical about the plane of the moon's orbit. Therefore, the declination can be measured either above or below the plane. Figures 4-24, 4-25, and 4-26 can be used to determine the first approximation of the launch azimuth, parking orbit coast angle, and injection velocity for a reentry angle of -6.4 degrees and a specified entry time and inclination of the geocentric conic.

PREPARED BY:	NORTH AMERICAN AVIATION, INC. SPACE and INFORMATION SYSTEMS DIVISION Figure 4-24	PAGE NO. 65 OF
CHECKED BY:		REPORT NO. SID 62-338
DATE:	TRANSEARTH TRANSIT TIME	[redacted] MODEL NO.



[redacted]

PREPARED BY:

CHECKED BY:

DATE:

NORTH AMERICAN AVIATION, INC.
SPACE and INFORMATION SYSTEMS DIVISION

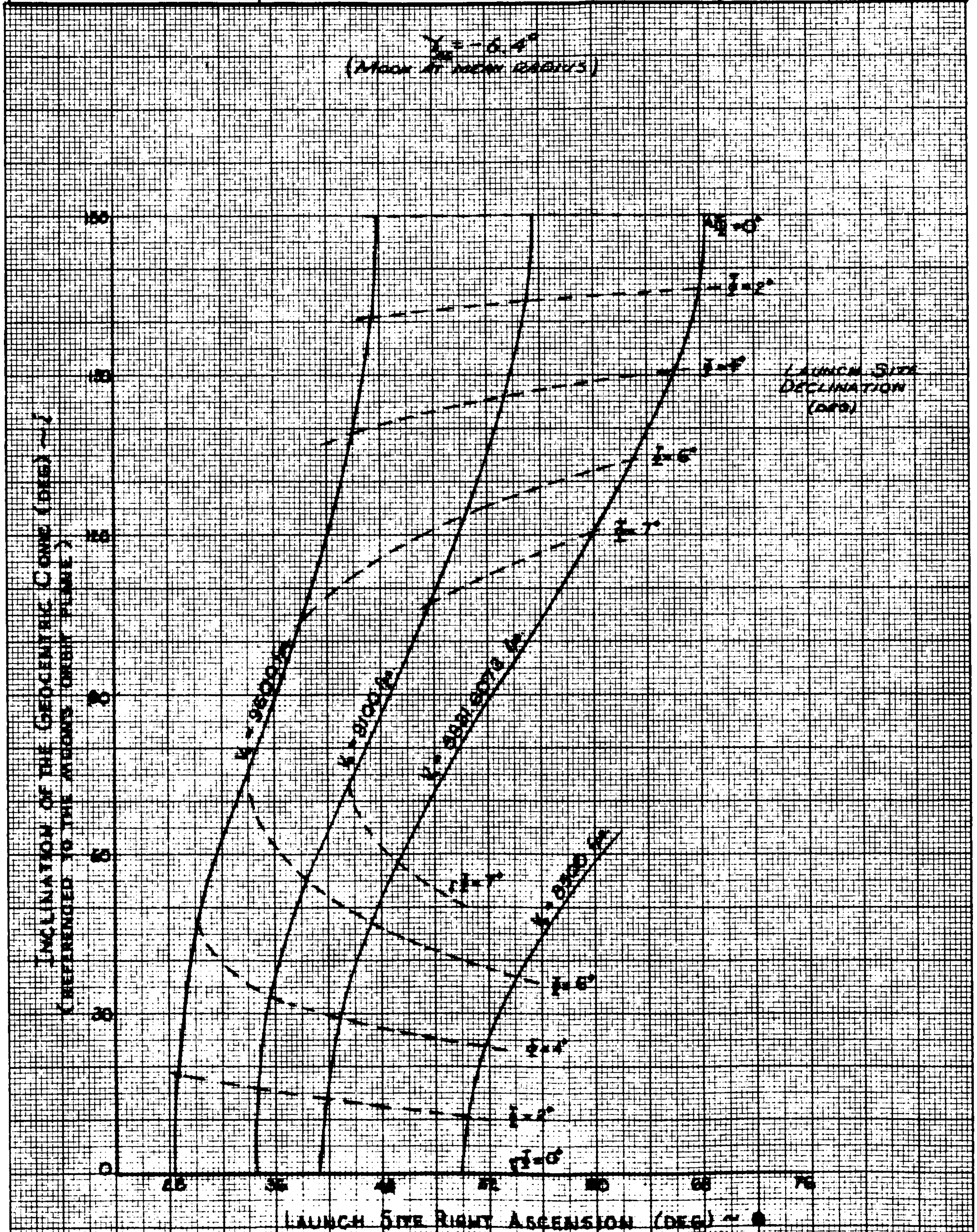
Figure 4-25

VERTICAL BOOST LAUNCH PARAMETERS

REPORT NO. SID 62-338

~~CONFIDENTIAL~~

MODEL NO.



~~CONFIDENTIAL~~

PREPARED BY:

CHECKED BY:

DATE:

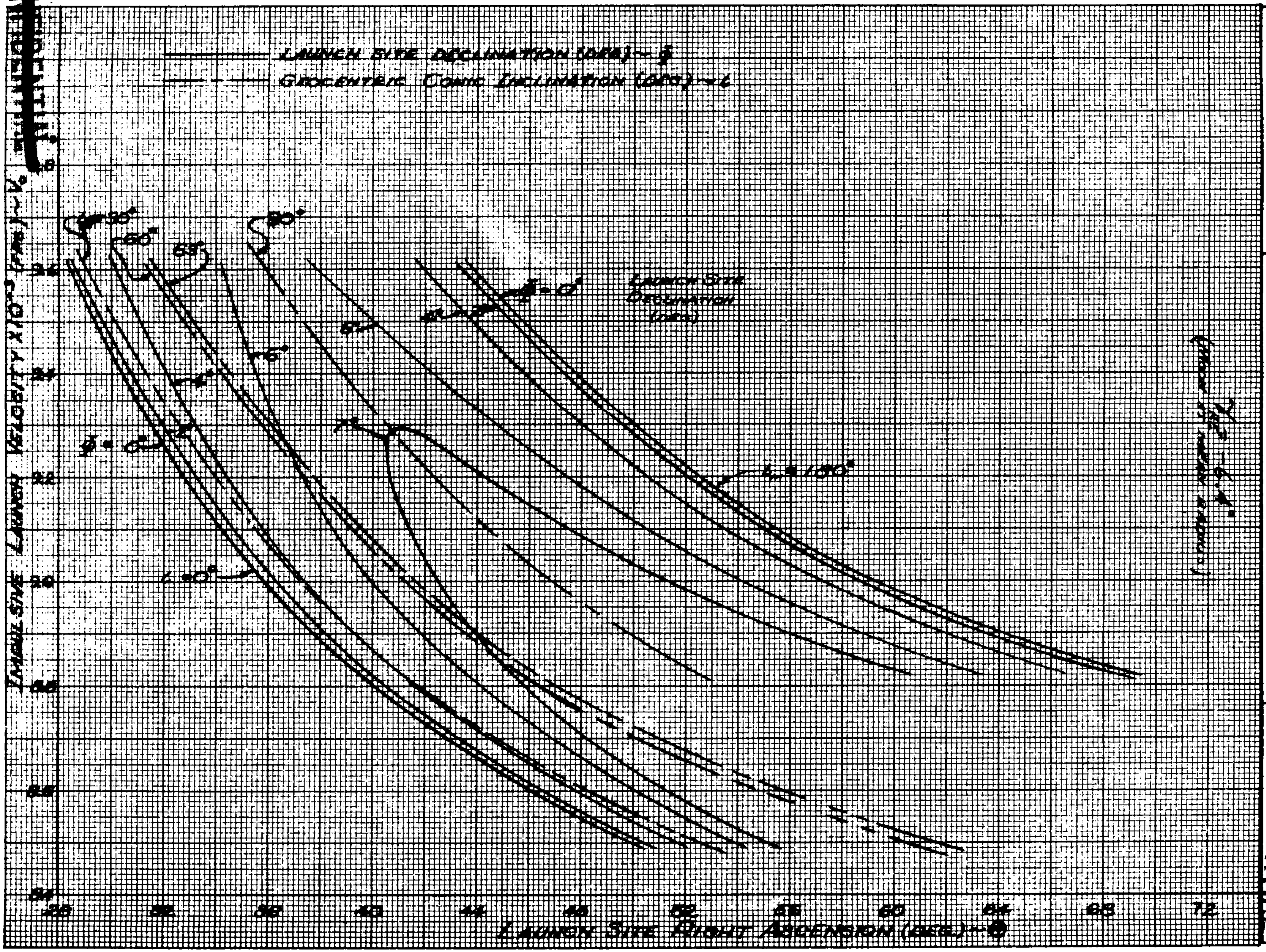
NORTH AMERICAN AVIATION, INC.
SPACE and INFORMATION SYSTEMS DIVISION

Figure 4-26

VERTICAL BOOST LAUNCH PARAMETERS

REPORT NO. SID 62-338

MODEL NO. ~~CONFIDENTIAL~~



~~CONFIDENTIAL~~

The nominal velocity at injection into the transearth trajectory for the lunar landing mission is to be 8,800 ft/sec. A lunar launch delay of approximately four hours can be obtained by using the established inclination of the geocentric conic by nominally launching so that the entry will be along the eastern boundary of the GOSS track. As the launch is delayed, the inclination is decreased reaching a minimum 29.48 degrees after a two hour delay. Further launch delays require an inclination increase to the western boundary of the GOSS track. The entry range angle required is a minimum when entering along the eastern boundary. Launch delays require an increase in the entry range angle reaching a maximum at the end of the launch window. If the entry range angle flexibility is not sufficient, the effective launch window will be decreased. Immediate launch from the lunar surface can be accomplished by reducing the velocity at injection into the transearth trajectory. The velocity decrease required is a function of the locations of the alternate earth landing sites available and the allowable inclination variation of the geocentric conic. The velocity at injection at 100,000 ft. must be reduced to as low as 8,430 ft/sec if the alternate landing sites specified for the earth orbit missions are used and the inclination restrictions are the same. The minimum velocity requirement is dictated by launches during those times of the month that the landing is in the northern hemisphere. It is believed that this velocity is approaching the minimum acceptable injection velocity. Further analyses, including the effects of eccentricity of the lunar orbit, are required to establish the minimum velocity. The minimum velocity required when returning to the southern hemisphere is 8,530 ft/sec at 100,000 ft. The increase is due to the availability of an additional alternate landing site. The injection velocity flexibility can be reduced by relaxing the geocentric conic inclination restraints or by increasing the number of alternate earth landing sites. The injection velocity flexibility must be increased, however, if the entry range angle flexibility is restricted.

4.2.4.3.2.2.2 Coast. - Same as 4.2.3.2.2.2.

4.2.4.3.2.3 Terminal Phase. - Same as 4.2.1.4.

4.3 ABORT REQUIREMENTS

4.3.1 General. - The integrated abort system shall provide an abort capability for all mission segments. This abort capability will include those situations in which the abort mode may be similar to or identical with the normal flight mode.

4.3.2 Typical Trajectories. -

4.3.2.1 Earth Orbital Mission. -

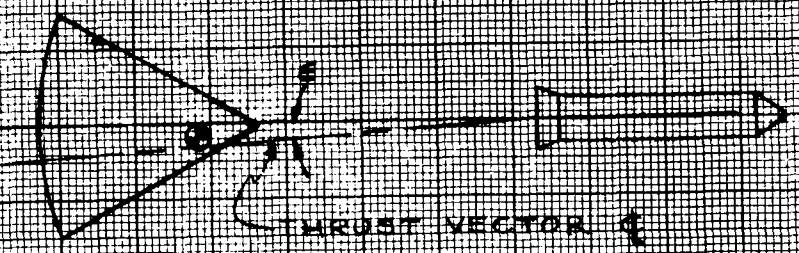
4.3.2.1.1 Ascent. - Ascent aborts are classified as atmospheric and extra-atmospheric aborts. Atmospheric abort is accomplished by separating the command module using a solid-propellant dual-propulsion system. The launch escape system (LES) shall separate the command module from the launch vehicle in the event of failure of the launch vehicle. The LES shall provide an abort capability through first stage boost and for the first 5 seconds of second stage firing, after which it shall be jettisoned. For pad aborts, the command module shall be propelled to a minimum altitude of 5,000 feet and a lateral distance at touchdown of at least 3,000 feet without exceeding emergency crew tolerances or structural limits. Figure 4-27 illustrates typical pad abort trajectories for various thrust offset angles and no control system. For midboost aborts, the command module shall be propelled a safe distance from a thrusting launch vehicle consonant with crew tolerances. Figure 4-28 presents typical time histories of the axial acceleration experienced during three abort conditions. When the atmospheric abort system is used, the spacecraft crew have little or no control in selection of the landing site. Depending on the time of abort, the abort range can vary between a few miles for pad abort to approximately 900 nautical miles for abort just prior to escape tower jettison. Therefore the possibility of recovery anywhere within this range must be considered. The escape tower is jettisoned shortly after second stage ignition and the extra-atmospheric abort is accomplished using the propulsion system (in the service module). The available characteristic velocity (2305 fps) was set by the orbital transfer, orbital ejection, and entry requirements. Of primary importance during extra-atmospheric abort (and also during high altitude atmospheric abort) are the g loads experienced by the spacecraft crew during atmospheric entry. Figure 4-29 shows the boost abort envelope. Also shown are the apogee limits for 10- and 20- g entry trajectories. The effects of the use of the Propulsion System were not included. Since the C-1 is payload-limited for the earth orbital mission, the abort situation encountered during C-1 boost establishes the most severe down-range landing requirements. Tentatively, four emergency landing sites will be necessary to assure continuous site selection during second stage burning.

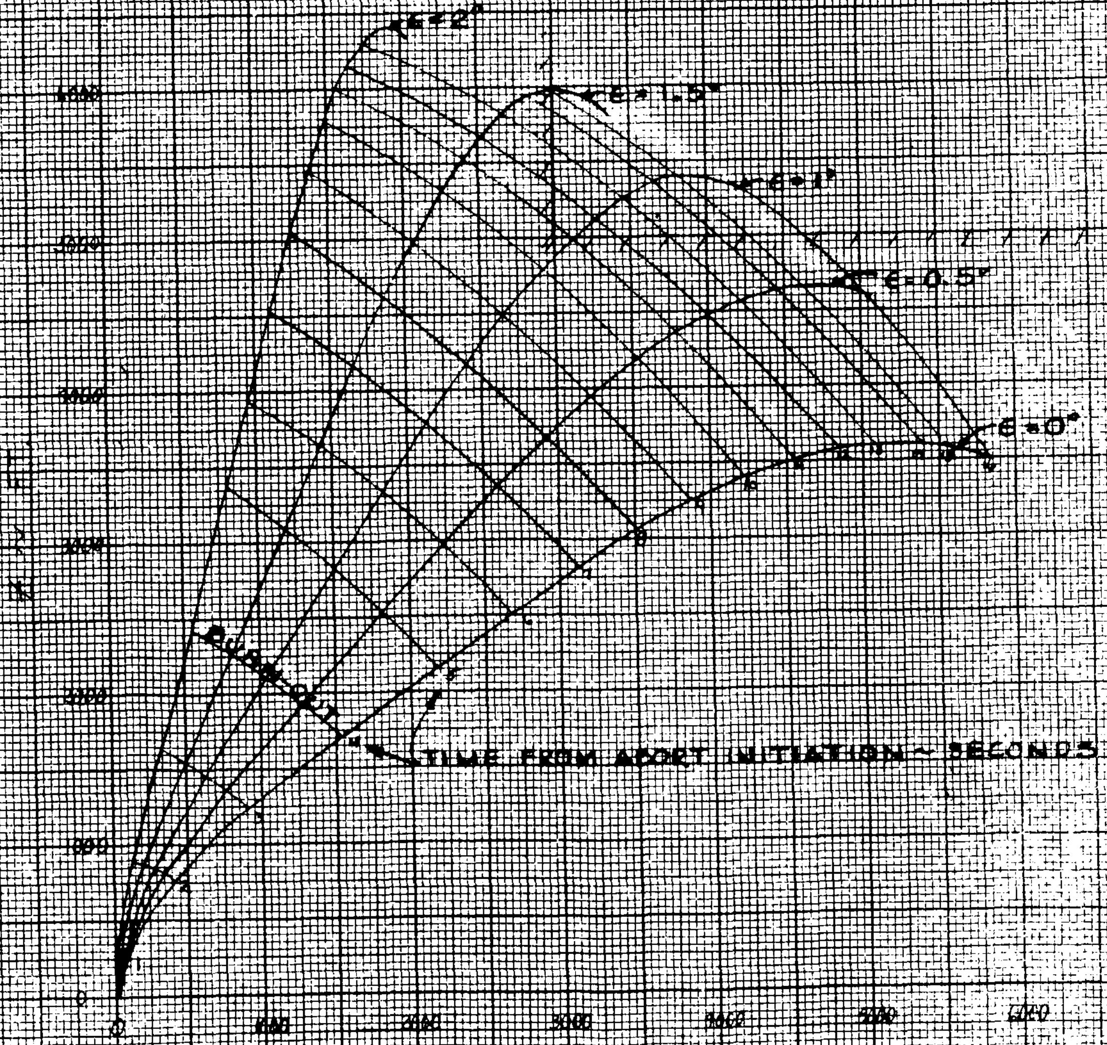
4.3.2.1.2 Orbit. - The dynamic requirements for abort from orbit are identical to the ejection requirements 4.2.1.3. The geometric requirements are altered only by the possible use of one of six alternate landing sites to permit abort from any orbital pass around the earth. The geographic location of the primary and alternate landing sites are as follows: (Table 4).

Some eastward deviation of the site locations in the northern hemisphere (westward deviation in the southern hemisphere) is permissible due to an overlap of landing coverage, providing the longitudinal spacing between adjacent sites does not exceed 60 degrees. However, no deviation in latitude is permissible without an increase in the required dogleg at orbital ejection. (See paragraph 4.2.1.3)

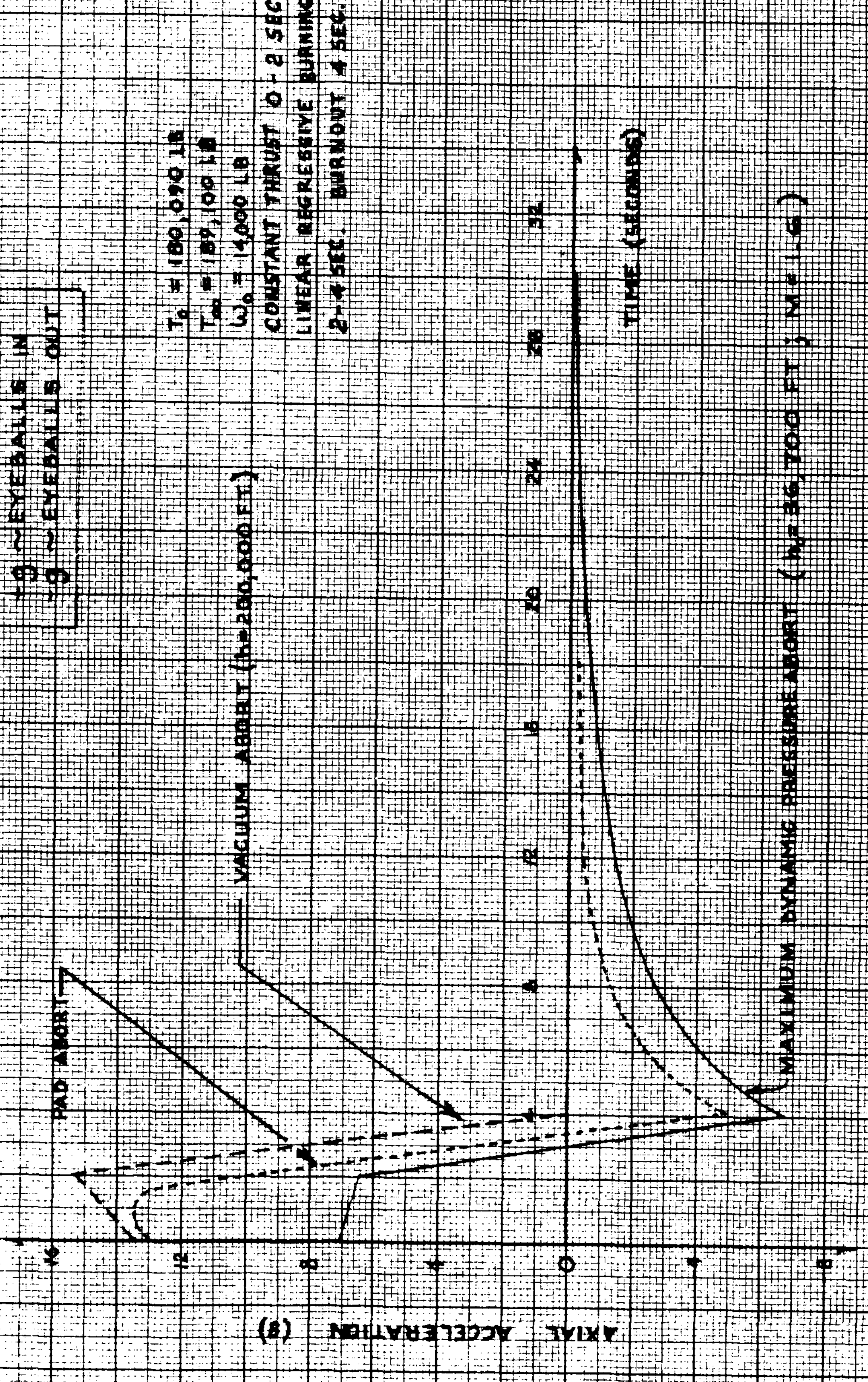
Figure 4-27

TYPICAL PAD ABORT TRAJECTORIES





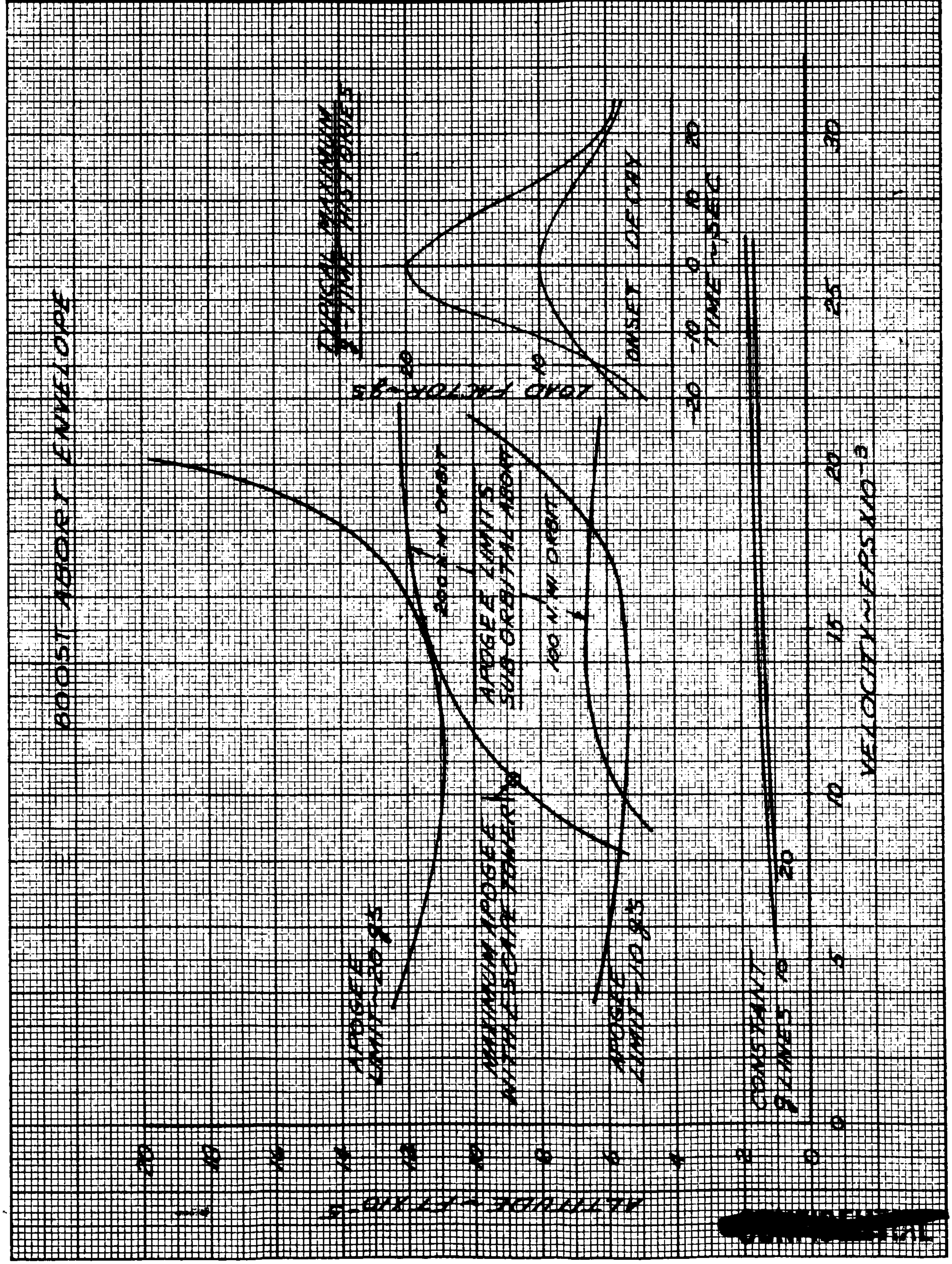
PREPARED BY:	NORTH AMERICAN AVIATION, INC. SPACE and INFORMATION SYSTEMS DIVISION	PAGE NO. 71 OF
CHECKED BY:		REPORT NO. SID 62-338
DATE: [redacted]	Figure 4-28	MODEL NO. ~~CONFIDENTIAL~~



TYPICAL AXIAL ACCELERATION TIME HISTORIES

~~CONFIDENTIAL~~

Figure 4-29



[redacted]

Table 4. Landing Site Locations

Site No.	Latitude	Longitude	Approximate Location
1	29.48 N	99.00° W	San Antonio, Texas (primary)
2	29.48 N	159.00° W	Hawaiian Islands
3	29.48 N	141.00 E	Japan
4	-29.48 S	45.00 W	Brazil
5	-29.48 S	105.00 W	Kermadic Islands
6	-29.48 S	165.00 W	Easter Island
7	-29.48 S	135.00 W	Woomera, Australia

4.3.2.2 Lunar Missions. -

4.3.2.2.1 Translunar. - Considerable data must be generated to establish properly the requirements for translunar abort within the framework of the following three classes of emergency:

Class I: Minimum-time to return to the earth. A Class I abort would be due to a catastrophic-type system failure, such as life support system, guidance, etc. Maximum stress entry conditions are required.

Class II: Short time return to a specific landing site using unrestricted landing azimuth. This type abort implies urgency and may result from a system failure that has a predictable decay rate. This requires only a single powered maneuver at the point of abort.

Class III: Normal abort to a specific landing site through a specified entry track. This is a controlled two-impulse maneuver abort and may be the result of mission cancellation, impending solar flares, excessive insertion errors, etc.

4.3.2.2.1.1 Initial Phase. - Abort during the ascent phase is similar to abort from the ascent phase of the orbital mission, except that additional performance capability is available from the lunar mission configurations. Since the launch azimuth will vary as a function of launch time, lateral maneuvering will probably prove necessary during boost abort. Tentatively only one downrange site will be required for successful abort during lunar mission boosts. Abort from the parking orbit is similar in aspect to ejection from orbit during the earth orbital mission with the exception that more propellant is available for planar change to land at a chosen site. As insertion boost progresses, the capability of abort to a pre-selected landing site becomes more difficult, and at insertion the flexibility is at a minimum.

Figure 4-30 is presented to indicate the choice of coplanar abort range, and abort time as a function of characteristic velocity applied at insertion. It may be seen that for the fully loaded capability of 10,335 fps in the SM, an abort time under 1 hour could be achieved.

4.3.2.2.1.2 Midcourse Phase. - If an emergency occurs after translunar injection and abort can be delayed some 16 minutes, the spacecraft will have coasted to a flight path angle sufficiently high to permit a normal abort. A typical translunar flight history may be used for reference. In general, though, no firm velocity requirement for mid-course abort can be established until an extensive parametric study of the 3 classes of abort is made. For a typical example of Class II abort, Figure 4-31 presents the characteristic velocity requirements as a function of abort radius and desired date of return. Figure 4-31 indicates that in order to return to the earth in two days (abort for solar flares), a velocity increment of about 10,000 fps is required.

4.3.2.2.2 Lunar Vicinity. -

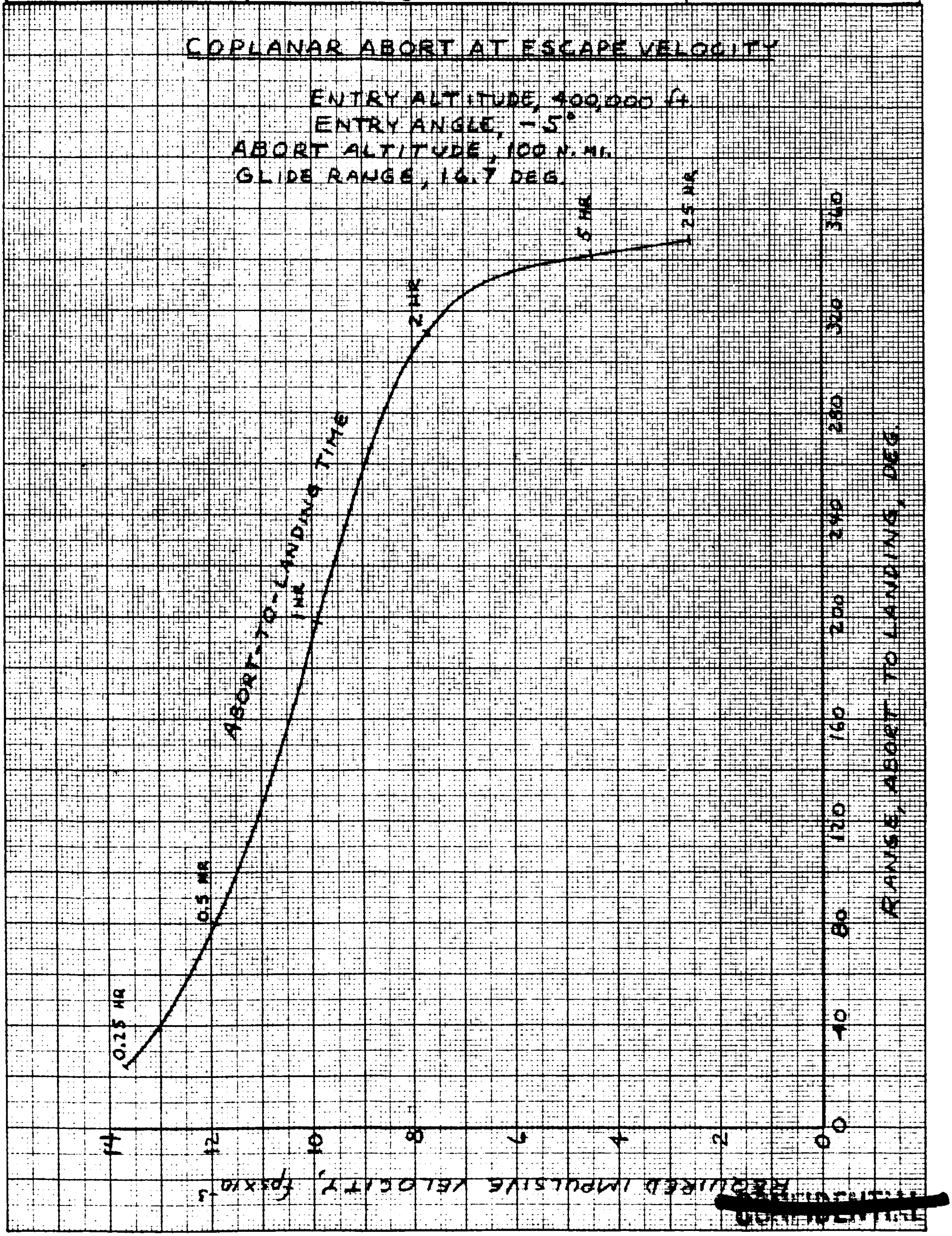
4.3.2.2.2.1 Circumlunar. - For all lunar missions the spacecraft propellant tanks will be fully fueled. Assuming that 1000 fps is required for translunar and transearth guidance corrections, 9,335 fps characteristic velocity remains on board for abort. (See Table 2.) Part of this velocity increment could be used to reduce transearth transit time by as much as 24 hours. The residual fuel would then be used to slow to an elliptical entry velocity. Depending on the class of abort, no difficulty should exist in reaching a preselected landing site.

4.3.2.2.2.2 Lunar Orbit. - The most severe performance requirements for the lunar orbit mission dictate an available characteristic velocity of 9200 fps. For this mission, therefore, as little as 1335 fps may remain aboard for reducing the transearth transit time after injection.

4.3.2.2.2.3 Lunar Landing. - For the lunar landing mission an abort capability is maintained up to touchdown. Should abort prove necessary during the landing phase, only a small vertical velocity need be negated by the Propulsion System (in the service module). Should abort prove necessary during main retro, the Propulsion System can provide the required acceleration to pull up safely. For this type of abort, reserve fuel will always remain at earth entry if a nominal transearth injection velocity of 8800 fps is used.

4.3.2.2.3 Transearth. - Abort during transearth flight restricts itself to reducing the transit time. For the circumlunar missions the maximum capability to reduce transit time exists while for the lunar landing mission

Figure 4-30



COPLANAR ABORT AT ESCAPE VELOCITY

ENTRY ALTITUDE, 400,000 ft
ENTRY ANGLE, -5°
ABORT ALTITUDE, 100 N. MI.
GLIDE RANGE, 16.7 DEG.

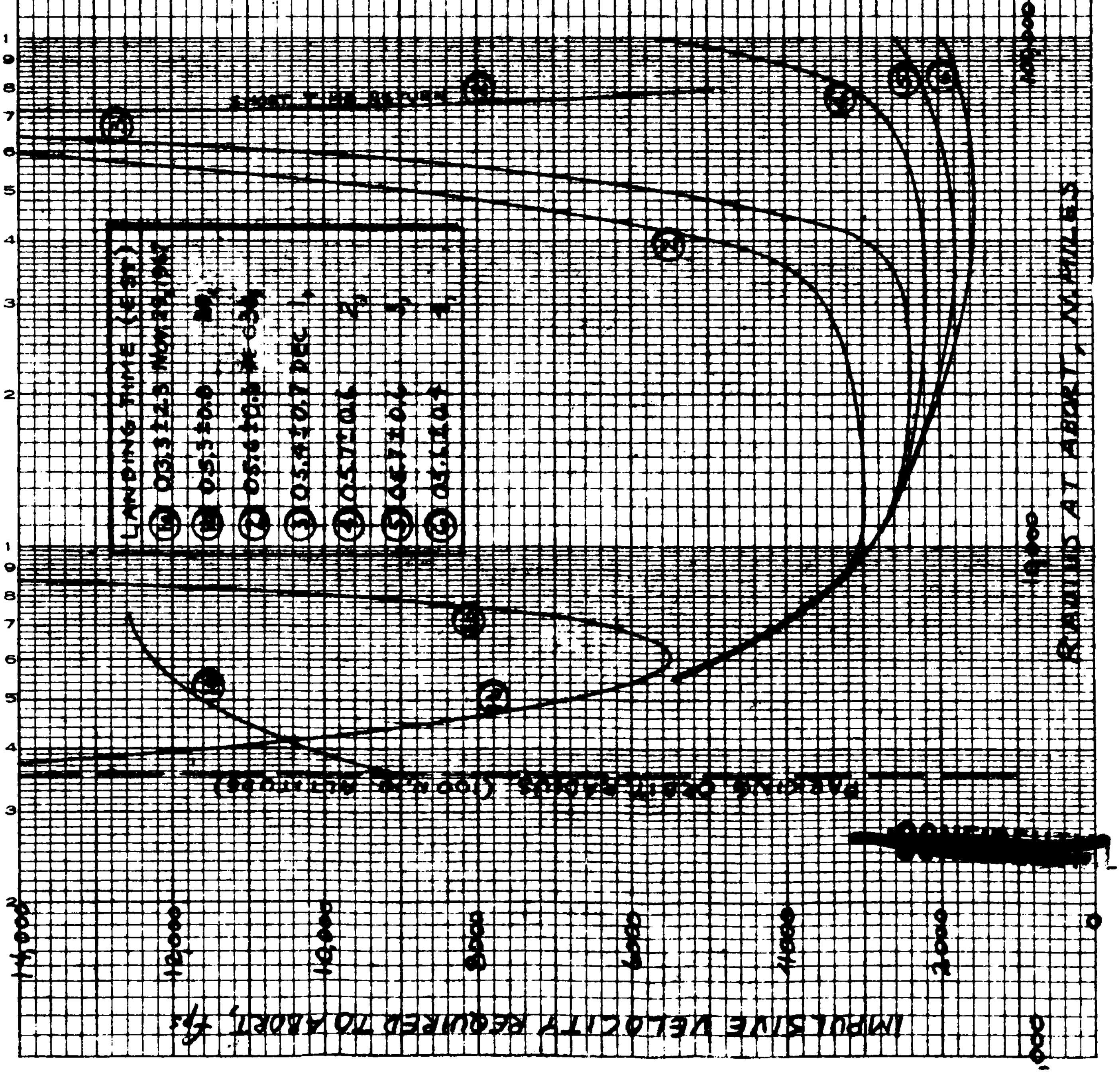
[redacted]

Figure 4-31

TYPICAL CLASS II ABORT IMPULSIVE VELOCITY REQUIREMENTS

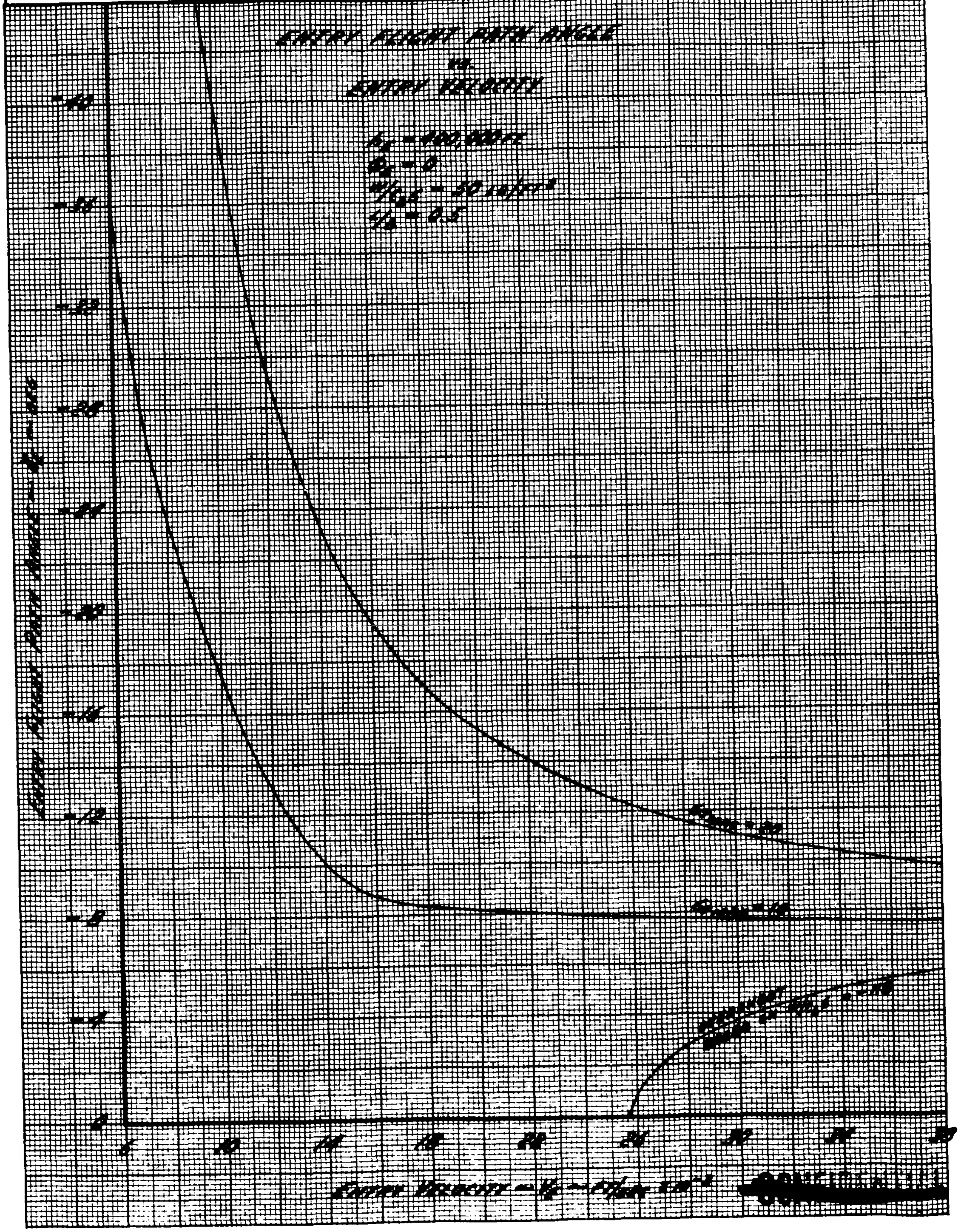
CONFIDENTIAL

LUNAR MISSION INSERTION DATE, 23.92 HRS, NOV. 28, 1967
LANDING SITE, SAN ANTONIO, TEXAS
ENTRY ALTITUDE, 400,000 ft
ENTRY FLIGHT PATH ANGLE, -5°
GLIDE RANGE, 16.7°



no transit time flexibility remains if the nominal flight plan is followed. Figure 4-32 shows the entry flight path angle limits as a function of the entry velocity that may result from an aborted mission.

Figure 4-32



[redacted]

5.0 DESIGN TRAJECTORY CRITERIA

5.1 Nominal Trajectories. - The Spacecraft design shall be compatible with all mission requirements through lunar landing with the C-1, C-5 and NOVA class launch vehicles. Nominal trajectories and trajectory envelopes shall be established for all missions based on flight plan design criteria. These trajectories will form the basis for defining systems characteristics. They shall further aid in identifying that flight phase and missions most critical to each spacecraft system and subsystem. The basic data, assumed to obtain the nominal trajectories, shall consist of design or expected performance levels for all systems and designated standard environmental conditions.

5.2 Off-Design Trajectories. - Off-design trajectories and trajectory envelopes shall form the basis of final spacecraft systems performance criteria and guaranteed mission performance. For each mission, flight phase, and configuration, deviations from the nominal or standard conditions in all items that affect trajectory characteristics shall be assessed. These items consist primarily of the following:

(1) Aerodynamic and structural tolerances

(2) Propulsion tolerances

(3) Guidance and control tolerances

(4) Environmental criteria.

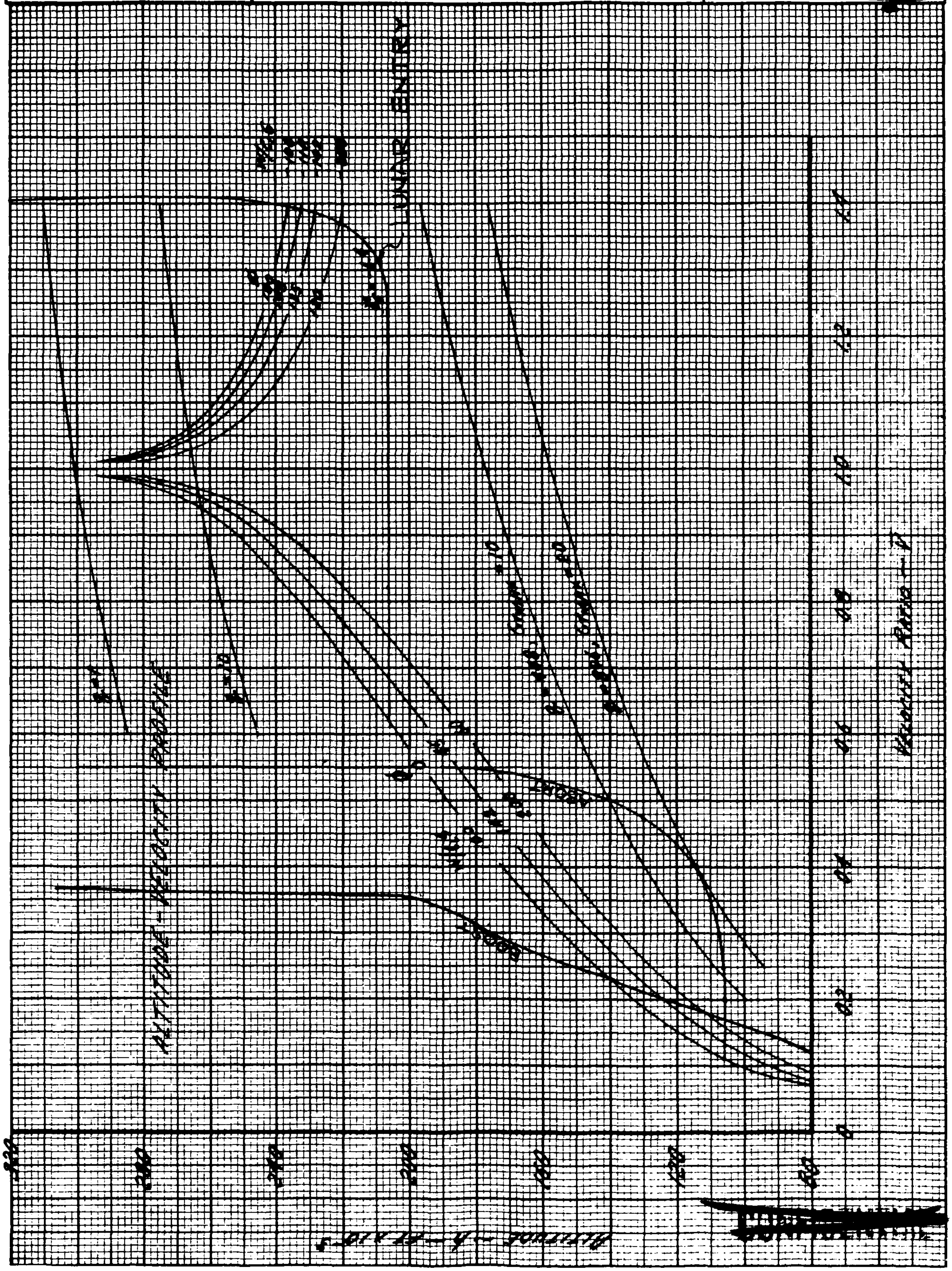
Assessment of tolerances and expected deviations shall be consistent with overall systems reliability requirements to meet the specified mission success and crew survival probabilities.

5.3 Atmospheric Flight Phases. - Atmospheric flight phases include boost, entry and recovery, and aborted flights. Altitude-velocity characteristics for these phases are illustrated in Figure 5-1.

5.3.1 Boost. - The boost trajectory shall be based upon the launch vehicle weights and aerodynamic coefficients indicated in the Spacecraft Launch Vehicle Integration Report and the payload weights indicated in the Spacecraft Description Manuals. Boost trajectories shall include the effects of: a rotating, oblate earth; launch azimuth; and, a standard wind profile and atmosphere. Altitude variations of pressure, temperature, and density will be as recorded in NASA TN D-595, "A Reference Atmosphere for Patrick AFB, Florida".

To obtain maximum payload, boost trajectories will steer a zero lift trajectory until reaching a dynamic pressure of 10 pounds per square foot. Thereafter, an optimum altitude steering program will be used to optimize

PREPARED BY: LH	**NORTH AMERICAN AVIATION, INC.** **MISSILE DIVISION**	PAGE NO. 80 OF
CHECKED BY:		REPORT NO. SID 62-338
DATE:	Figure 5-1	[redacted] MODEL NO.



[redacted]

payload into different burnout altitudes and velocities. Between ignition and burnout of a stage, there is usually lost overboard lubrication fluid or hydraulic fluid, chill down, and pressurization gases. For trajectory computation, the estimated values of these items will be assumed to be discharged overboard at a constant rate during stage burning. The exception to these criteria will be made for items expended specifically during separation of stages.

Except for special analyses, the normal trajectories will not include thrust variations, buildup or cutoff impulse variations or mixture ratio deviations. Sufficient propellant shall be held in reserve in each stage for expected variations in mixture ratio. The engine manufacturers nominal specific impulse shall be utilized to determine propellant flow rates and normal engine burning time.

5.3.1.1 C-1 Launch Vehicle. The primary mission of the C-1 launch vehicle is to carry a payload into a circular earth orbit. A secondary mission is to carry a reduced payload to an atmospheric entry condition at greater than orbital velocity. A direct mode of ascent shall be considered since the second stage has no coast attitude control system for engine restart.

The capability of both stages to continue flight with one engine failing enhances considerably the mission reliability of the vehicle. Alternate missions, or trajectories, shall be computed at reduced performance reflecting failure of one engine.

The trajectory shall include the effects of the separation procedure. For the C-1, the four inner engines will cut off 6 seconds prior to the outboard engines. Ignition of the second stage is assumed to occur 1.7 seconds after final cutoff of the first stage.

5.3.1.1.1 S-I Stage.- Trajectory performance of the S-I stage shall include the effects of propellant depletion during a 3 second mainstage holddown prior to vehicle release. Axial thrust of the S-I stage shall be based upon 4 engines canted at 3 degrees and 4 engines canted at 6 degrees.

5.3.1.1.2 S-IV Stage. - Axial thrust of the S-IV stage shall be based upon 6 engines canted at 6 degrees.

5.3.1.2 C-5 Launch Vehicle. - The primary mission of the C-5 launch vehicle is to carry payloads into circular earth orbits or to earth escape velocities. Direct and indirect (Hohmann) modes of ascent shall be considered; however, trajectories shall reflect the influence of no restart capability for the S-II stage. The first and second stages of the C-5 launch vehicle will have the capability to continue flight with one engine failing. Alternate trajectories shall be computed for this consideration.

5.3.1.2.1 S-IC Stage. - The trajectory criteria related specifically to this stage will be supplied when specific design details are available.

5.3.1.2.2 S-II Stage. - The trajectory criteria related specifically to this stage will be supplied when specific design details are available.

5.3.1.2.3 S-IVB Stage. - (Refer to 5.3.1.2.2)

5.3.2 Entry. - The entry trajectory consists of that portion of the mission between the point where the vehicle first reaches an altitude of 400,000 feet (entry interface) and the point where the altitude has finally been reduced to 100,000 feet (recovery interface). The entry configuration corresponds to the command module. The vehicle's path is controlled aerodynamically by rotating the lift-force vector about the velocity vector. A fixed aerodynamic trim angle of attack is achieved by offsetting the center of gravity. The trim condition is chosen on the basis of achieving a maximum lift-drag ratio of 0.5.

The command module systems shall be designed to enter the atmosphere at all speed conditions up to escape speed at specified entry path angle limits without exceeding a 20 g load factor. The critical entry and subsequent flight profile however corresponds to lunar missions. All other missions, including aborted missions and survival flight profiles shall be defined for establishing MCC, GOSS requirements and special criteria such as displays (for example).

5.3.2.1 Entry Corridor - Lunar Missions. - The entry corridor depth shall be defined as the difference in conic (vacuum) perigee altitudes between the overshoot and undershoot boundaries. It shall also be defined as the difference in minimum (overshoot) and maximum (undershoot) flight path angle at the interface altitude (400,000 feet) assuming local escape velocity. Several corridors are defined; i.e. (1) Design Corridor, (2) Operational Corridor, (3) Midcourse Guidance Corridor.

5.3.2.1.1 Design Corridor Criteria. - The design corridor depth and allowable entry conditions shall be determined on the basis of standard 1959 ARDC Atmosphere, spherical, non-rotating earth, design aerodynamics and design loading conditions. The corridor limits shall be based upon the initial entry (to pull-out) at positive L/D = 0.5. The Overshoot Boundary shall be defined by the minimum entry angle for which a constant altitude capability exists at the initial pull-out point of the trajectory when the vehicle is instantaneously rolled to 85 percent of its maximum available negative lift. The Undershoot Boundary shall be defined as that entry angle for which a 10g total load factor is obtained during the initial atmospheric penetration. The boundaries of the design corridor shall be used as the initial entry conditions for establishing Spacecraft systems design trajectories. A 20g entry condition, although not a part of the corridor, shall also be considered as a special load for lunar entries. This extreme case could occur in the event of an extreme guidance system error (for example) where the crew would still survive provided the systems worked satisfactorily. If this special load results in an appreciable design penalty, this criteria shall be re-examined with regard to its probability of occurrence.

5.3.2.1.2 Operational Corridor Limits. - The operational corridor boundary definitions are the same as those of the design corridor. The corridor limits however shall reflect the following items:

(1) 3σ atmospheric deviations and altitude profiles

(2) ±20% command module trim aerodynamics

(3) Pilot response characteristics (to be determined)

(4) Stabilization and control tolerances (to be determined)

(5) Guidance sensing tolerances (to be determined)

The boundaries of the operational corridor shall be used as the initial entry conditions for establishing guaranteed performance envelopes. Further, the operational corridor depth shall represent the absolute limits for midcourse guidance.

5.3.2.1.3 Midcourse Guidance Corridor. - The design objective for the midcourse guidance shall be eight (8) nautical miles. This corridor depth shall be used in establishing path control systems design criteria and touchdown dispersions for a specific range.

6.0 DESIGN TRAJECTORIES

6.1 Structural Loads and Heating Trajectories. - The design trajectories to be used for loads and thermal systems analyses are presented in Figures 6-1 through 6-4. These trajectories are limited to atmospheric flight and as such, represent maximum loads and load durations for both structural and aerodynamic heating analysis. Four trajectories define the design limits. These are : (1) Maximum range from the design corridor overshoot boundary, (2) Maximum range from the design corridor undershoot boundary, and (3) Minimum range from the design corridor undershoot boundary, and (4) the minimum range trajectory corresponding to a 20g entry condition. A summary of these trajectories is presented in Table 5. The trajectories are based upon the following conditions and assumptions:

(1) Initial entry conditions correspond to the Design Corridor boundaries,

(2) The vehicle flight mode is restricted to the atmosphere, i.e., maximum altitude following entry is 300,000 feet,

(3) The vehicle aerodynamics and loading correspond to $(L/D)_{max} = 0.5$ and $W/C_DA = 50$. The (L/D) is modulated by roll about the velocity vector. The entry maneuver until pull-out corresponds to a zero-degree bank angle,

(4) Standard 1959 ARDC Model Atmosphere,

(5) Initial conditions are based on 400,000 feet altitude. Terminal conditions are based on 100,000 feet altitude,

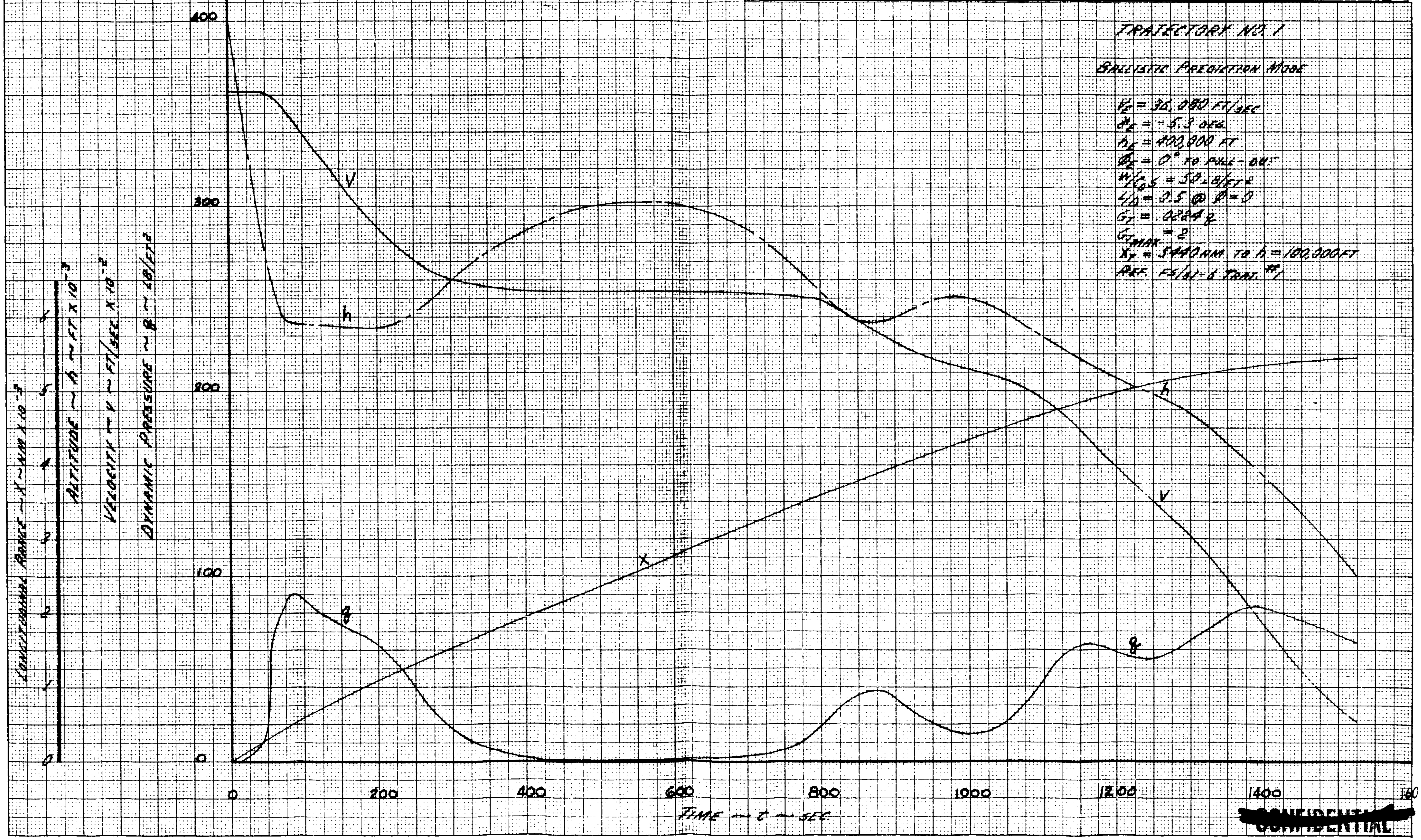
(6) And non-rotating spherical earth.

Table 5. Design Trajectories Summary

Traj. No.	γ EN Deg.	$\bar{q}$ MAX PSF	G_R MAX	$\dot{q}_s$ MAX BTU/FT^2-SEC	Q_S BTU/FT^2	R N. MILE	t SEC
1	-5.3	90	2	340	97,300	5440	1525
2	-7.4	450	10	600	64,700	3510	1100
3	-7.4	450	10	600	42,300	1157	464
4	-10.0	900	20			644	316

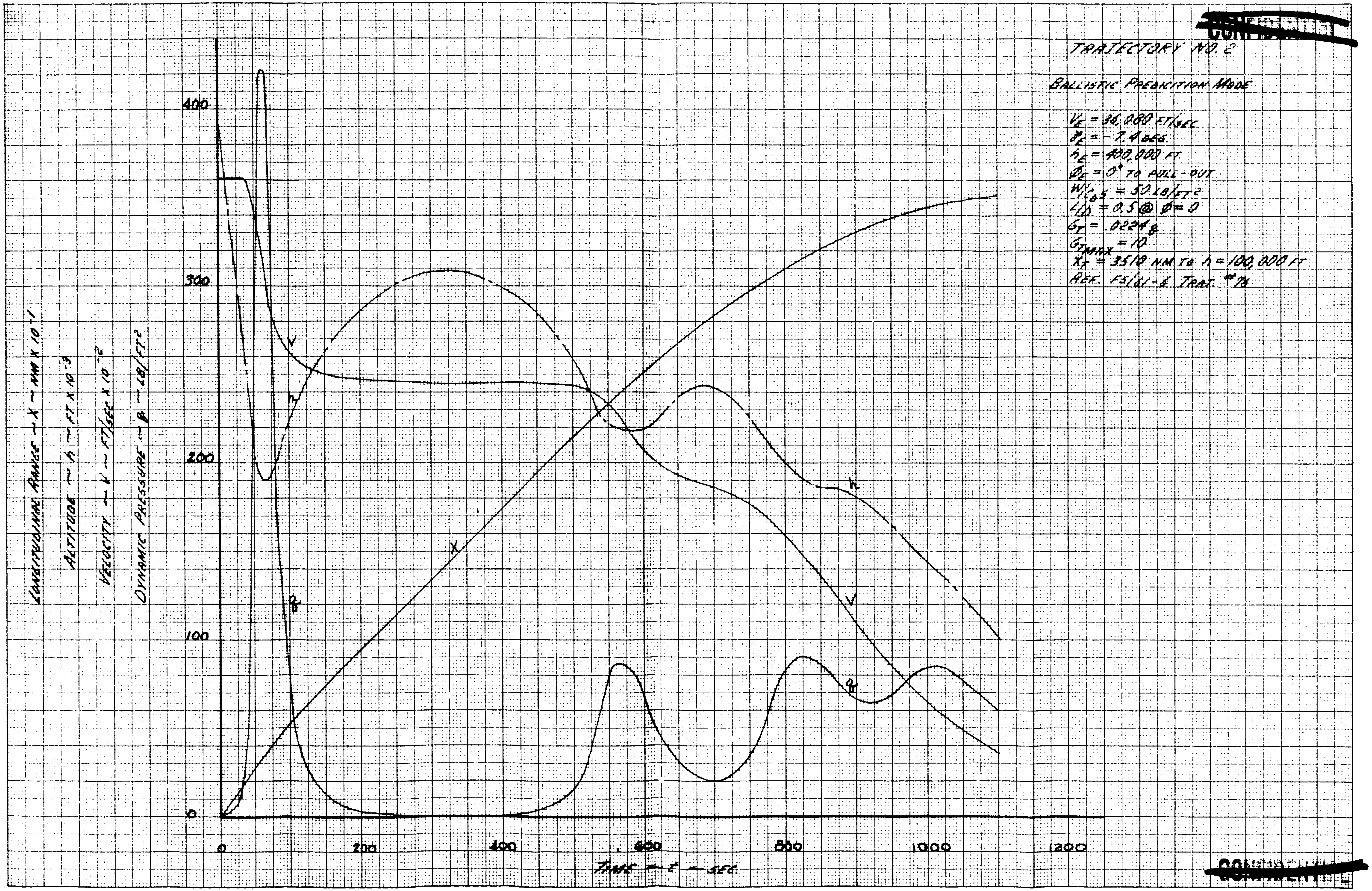
FORM M-113-N-3 REV. 9 58

PREPARED BY:	**NORTH AMERICAN AVIATION, INC.** **MISSILE DIVISION**	PAGE NO. 87. 88 OF
CHECKED BY:		REPORT NO. SID 62-338
DATE:	Figure 6-1	~~CONFIDENTIAL~~



~~CONFIDENTIAL~~

~~CONFIDENTIAL~~



TRAJECTORY NO. 2

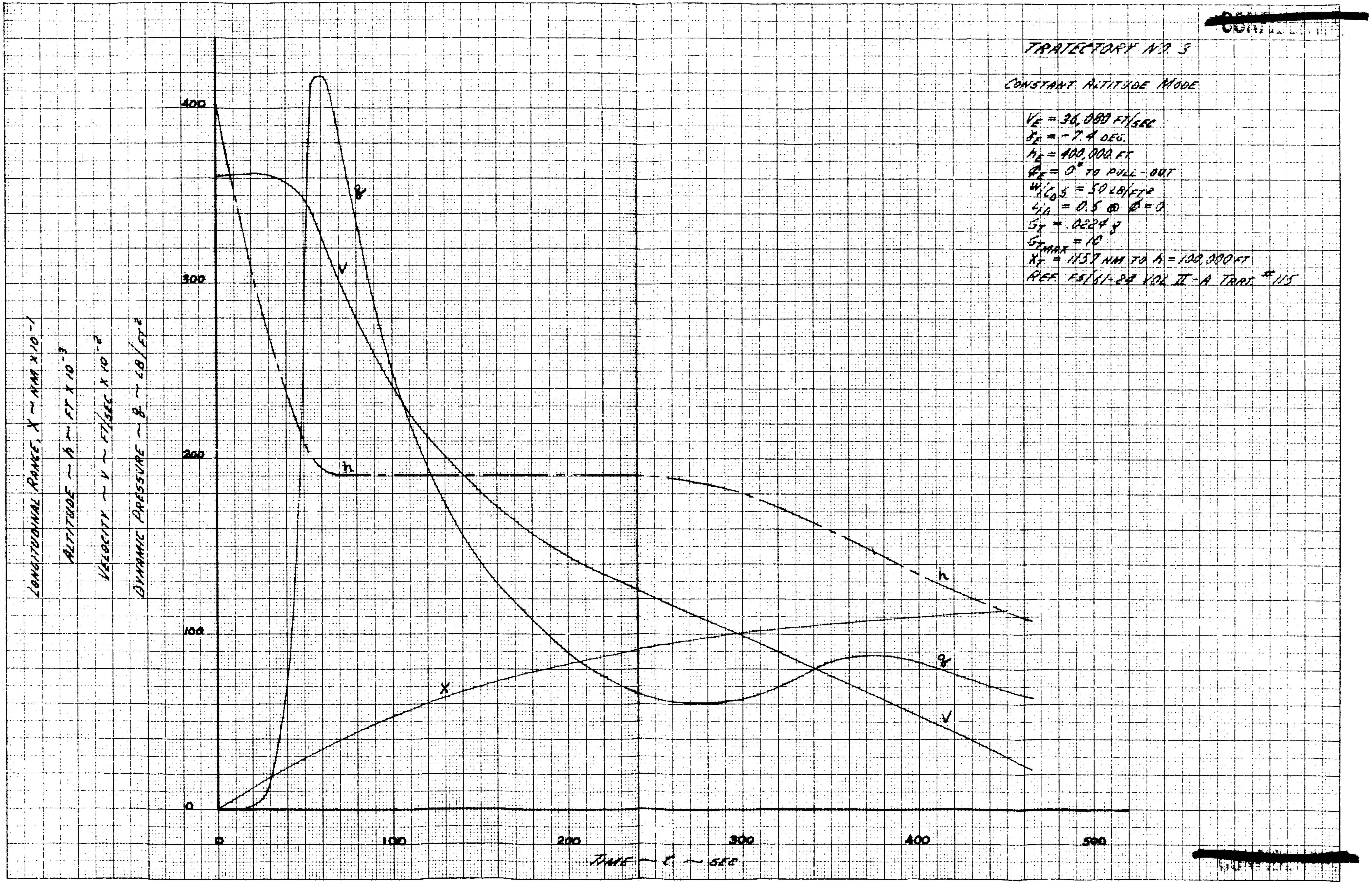
BALLISTIC PREDICTION MODE

V_E = 36,080 FT/SEC
γ_E = −7.4 DEG.
h_E = 400,000 FT
ϕ_E = 0° TO PULL-OUT
W/C_DS = 50 LB/FT²
L/D = 0.5 @ ϕ = 0
G_T = .0224 g
$G_{T_{MAX}}$ = 10
X_T = 3510 NM TO h = 100,000 FT
REF. FS/61-6 TRAJ. #76

~~CONFIDENTIAL~~

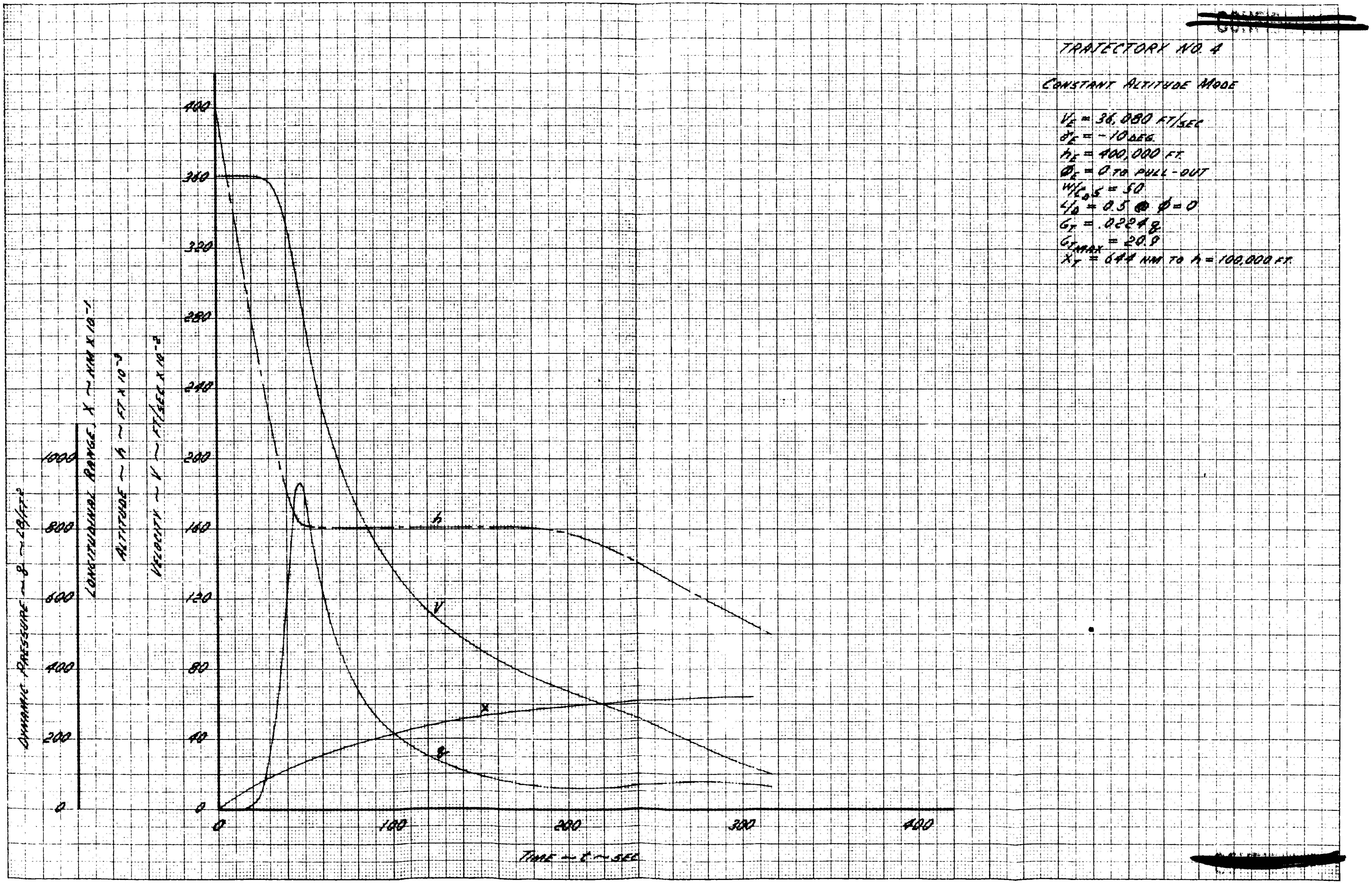
Figure 6-3

[redacted]



[redacted]

[redacted]



TRAJECTORY NO. 4

CONSTANT ALTITUDE MODE

$V_E = 36,080$ FT/SEC
$\gamma_E = -10$ DEG.
$h_E = 400,000$ FT.
$\phi_E = 0$ TO PULL-OUT
$W/C_D S = 50$
$L/D = 0.5$ @ $\phi = 0$
$G_T = .0224$ g
$G_{T_{MAX}} = 20.9$
$X_T = 644$ NM TO $h = 100,000$ FT.

[redacted]

~~CONFIDENTIAL~~

7.0 TRAJECTORY ENVELOPES

7.1 Mission Duration Envelopes. - Tables 6 through 9 summarize, by flight phase minimum, maximum and typical mission durations. Only a typical time is given where the min/max times are unknown. Service module burning times are based on a $T/W_0 = .477$ and landing module times reflect a $T/W_0 = .640$. Times shown are not associated with an abort situation.

Table 6. Earth Orbital Mission Duration

Phase	ΔTIME			Units	Remarks
	Min.	Typical	Max.		
Ascent	596	596	619	Sec.	C-1 (Full fuel load) 8 & 7 Engines
Orbit	1.08	23.9	335	Hrs.	Minimum, less than one orbit
Ejection	16.6	39.8	68	Sec.	
Coast		694		Sec.	Ejection to entry
Entry	180	516	2000	Sec.	600 to 7200 NM range
Recovery		663		Sec.	

Table 7. Circumlunar Mission Duration

Phase	ΔTIME			Units	Remarks
	Min.	Typical	Max.		
Ascent		552		Sec.	C-5, 3 Stages to Escape Velocity
Orbit	38	68	81	Min.	Required time is a function of launch date
Circumlunar Injection		234		Sec.	
Circumlunar Coast	138	139.99	139	Hrs.	Not free-return
Entry	930	601	1550	Sec.	3000 to 5500 n. mi. range
Recovery		663		Sec.	

~~CONFIDENTIAL~~

Table 8. Lunar Orbit Mission Duration

Phase	ΔTIME Min.	ΔTIME Typical	ΔTIME Max.	Units	Remarks
Earth Ascent		552		Sec.	C-5, 3 stages to escape velocity
Earth Orbit	38	68	81	Min.	Required time is a function of launch date
Translunar Injection		234		Sec.	
Translunar Coast	66	70.44	71	Hrs.	
Lunar Orbit Injection	163	180	183	Sec.	
Lunar Orbit	2	107.04	190.9	Hrs.	
Transearch Injection	182	242	242	Sec.	
Transearth Coast	59.5	66.3	71.5	Hrs.	
Entry	930	1368	1550	Sec.	3000 to 5500 NM range
Recovery		663		Sec.	

~~CONFIDENTIAL~~

Table 9. Lunar Landing Mission Duration

Phase	ΔTIME			Units	Remarks
	Min.	Typical	Max.		
Earth Ascent		505		Sec.	Typical NOVA
Earth Orbit	38	68	81	Min.	Required time is a function of launch date
Translunar Injection		360		Sec.	
Translunar Coast	66	70.44	71	Hrs.	
Lunar Orbit Injection		133		Sec.	
Lunar Orbit	1	1.89	3.1	Hrs.	
Hohmann Transfer		3540		Sec.	
Main Retro		180		Sec.	
Lunar Landing		188		Sec.	
Lunar Operations	0	111.41	175.89	Hrs.	May be 206.85 hrs., if all other times are minimum
Lunar Ascent		285		Sec.	
Lunar Orbit	0	1.58	2.1	Hrs.	
Transearth Injection	96	105	105	Sec.	8450/8800 fps.
Transearth Coast	59.5	62.35	80.2	Hrs.	
Entry	930	1368	1550	Sec.	3000 to 5500 NM range
Recovery		663		Sec.	

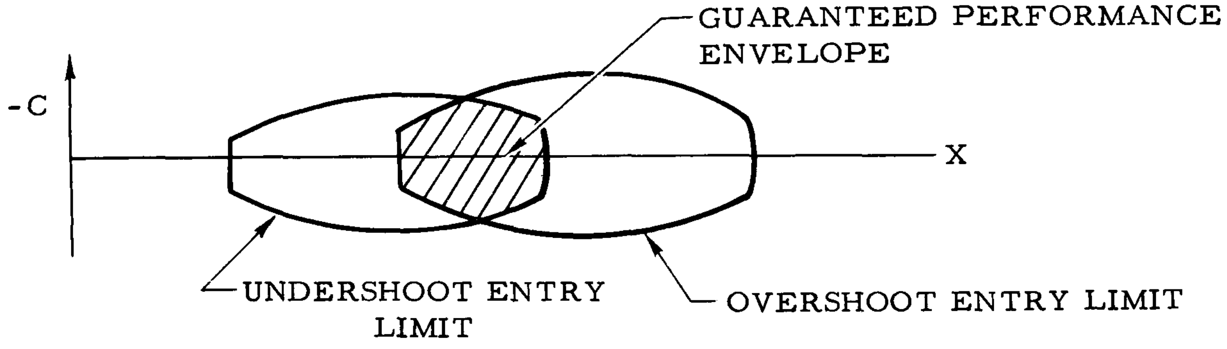
~~CONFIDENTIAL~~

7.2 Launch and Ascent Envelopes. - No data at this time.

7.3 Space Operation Envelopes. - No data at this time.

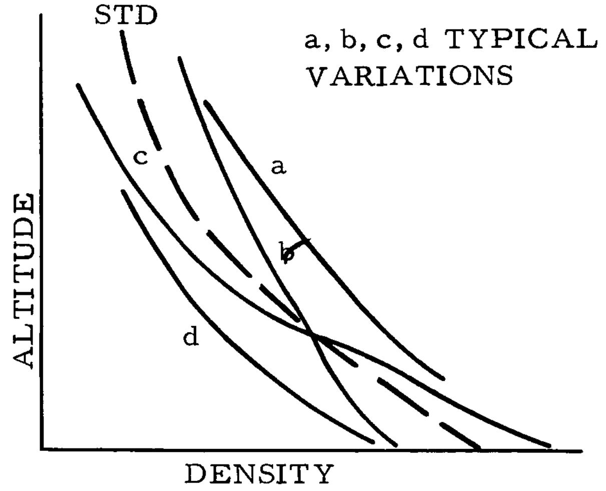
7.4 Entry Envelopes. -

7.4.1 Entry Range Envelopes. - Entry range envelopes (footprints) shall be established at the recovery interface based upon the overshoot and undershoot limits of the Operational Corridor. From these footprints the "guaranteed" performance envelopes shall be derived (see sketch) for (1) a flight plan restricted to the atmosphere and (2) a controlled "skip-out" flight plan.



The minimum guaranteed range shall correspond to the overshoot entry limit and human tolerance performance limit (10g maximum) constraints. The maximum guaranteed range shall correspond to the undershoot entry limit and the interrelations between trajectory sensitivity, guidance sensing accuracy and vehicle control capability. These footprints shall reflect the following conditions and assumptions:

1. Extreme density-altitude profiles. These profiles shall include geographical and seasonal variations encountered between ±34-degrees of latitude (see sketch)



2. The reduction of the trim L/D from a value of 0.5 to a value of______ at the initial pull-out point of the entry trajectory shall be derived from a statistical evaluation of the ablation effects on trim aerodynamics. The corresponding value of W/C_DA shall be______

3. The atmospheric exit and re-entry altitude shall be 300,000 feet the controlled "skip-out" mode. In addition, this altitude shall be the limit for atmospheric flight modes

4. The total integrated heat load resulting from the maximum range design trajectory, Figure 5-2, shall not be exceeded.